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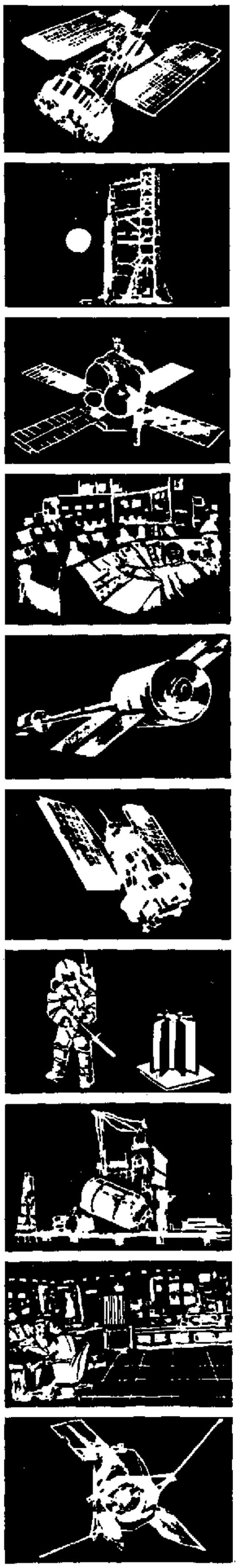

EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY

Report No. 6

SPACE SHUTTLE INTERFACES/UTILIZATION

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Prepared for:
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EARTH OBSERVATORY SATELLITE
SYSTEM DEFINITION STUDY

REPORT NO. 6

SPACE SHUTTLE
INTERFACES UTILIZATION

PREPARED FOR:
GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND 20771
UNDER
CONTRACT NO. NAS 5-20518

PREFACE

This report, "Space Shuttle Interfaces/Utilization," has been prepared for NASA/GSFC under contract NAS 5-20518, EOS System Definition Study. It describes the mechanical, electrical and operational interfaces between Space Shuttle and the EOS spacecraft in addition to recommending how Space Shuttle can be most economically utilized for the EOS program.

REPORT #6

SPACE SHUTTLE INTERFACES/UTILIZATION

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SECTION 1.0
SUMMARY

EOS is one of the first spacecraft being designed to be compatible with Space Shuttle. The investigations of the mechanical and electrical interfaces, the impact of Shuttle operations and the cost benefits accrued from Shuttle utilization therefore become key areas of interest in the EOS study. This report documents these investigations which have indicated that EOS can be made compatible with Space Shuttle and in the process significantly reduce program costs. The design definition of Shuttle is being currently available in preliminary form, thus requiring flexible implementation concepts for EOS. This is especially true in the electrical interface area where very little detail Shuttle information is available. Additional Shuttle interface areas requiring better definition are safety criteria, contamination and thermal control. These later areas are addressed in appendices to this report.

The MECHANICAL INTERFACES between EOS and Shuttle have had considerable emphasis since separate studies have been performed by RI and SPAR/DSMA to develop a Shuttle bay support system for EOS. This support system, shown in Figure 1-1, consists of:

- Large equipment storage fixture
- Launch/retrieve support cradle
- Docking frame and erection mechanism
- SPMS exchange mechanism
- Module storage magazine
- Shuttle attached manipulator system (currently baselined for the shuttle)

This full complement of mechanical support equipment may be used during a combined EOS delivery/retrieval and service mission. Reduced subsets of this equipment can be used for delivery or retrieve only missions. General Electric has used the RI and SPAR/DSMS hardware definitions in establishing the EOS mechanical interfaces with the Shuttle.

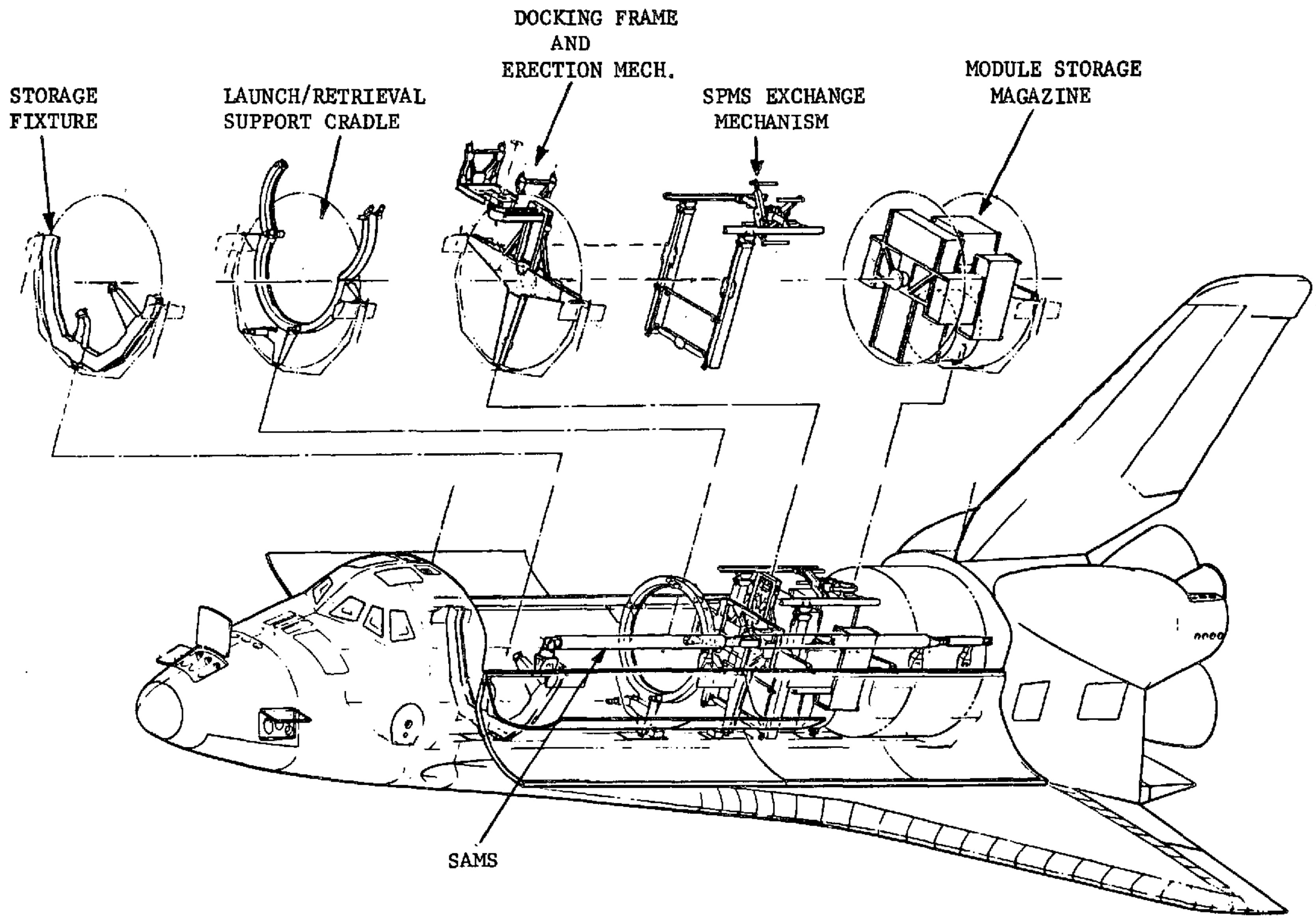


FIGURE 1-1 SHUTTLE EOS FLIGHT SUPPORT SYSTEM

The mechanical interfaces and provisions required for a launch or retrieval EOS mission are summarized in Figure 1-2. These features include mechanical interfaces for support of EOS in the Shuttle bay during Shuttle flight and provisions for retrieval or release of EOS by Shuttle. Other features required in the EOS design are provisions to refold appendages such as solar arrays, antennas and booms in addition to providing covers for critical equipment. The Shuttle flight support equipment required for this mode of operation consists of the:

- Launch/retrieve support
  (The design of this support has been simplified and the weight reduced by replacing the EOS transition ring with a three-point transition frame.)
- Docking frame and erection mechanism
  (The use of this equipment for the launch and retrieve mission is optional; SAMS may be used to place the spacecraft directly into the launch/retrieve support.)

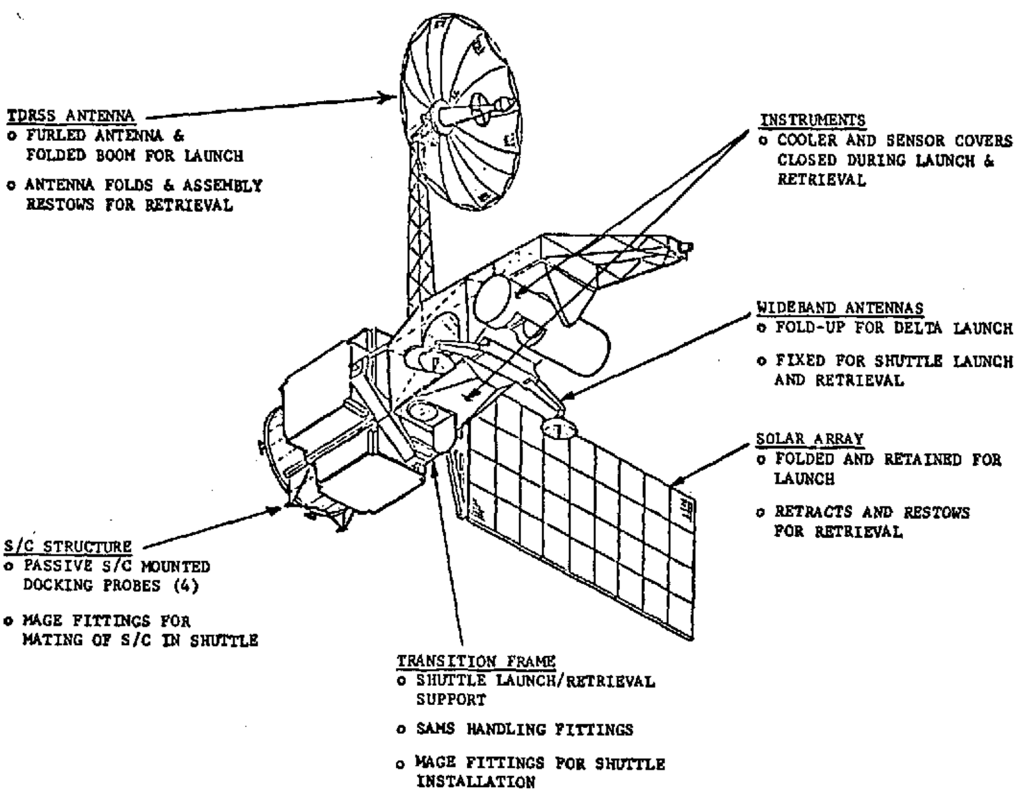


FIGURE 1-2 EOS SPACECRAFT LAUNCH/RETRIEVAL PROVISIONS

The mechanical provisions and interfaces required for a resupplyable EOS spacecraft are summarized in Figure 1-3. Features that must be added for the resupply mode include corner latches and remote connectors for exchange of modules in the shuttle bay. Modularization of the instruments is also required to facilitate their remote exchange. An alteration to the Shuttle attached docking frame and erection mechanism is also required to allow axial removal and replacement of the propulsion module. This requirement has been coordinated with R.I. and deemed compatible with their basic design.

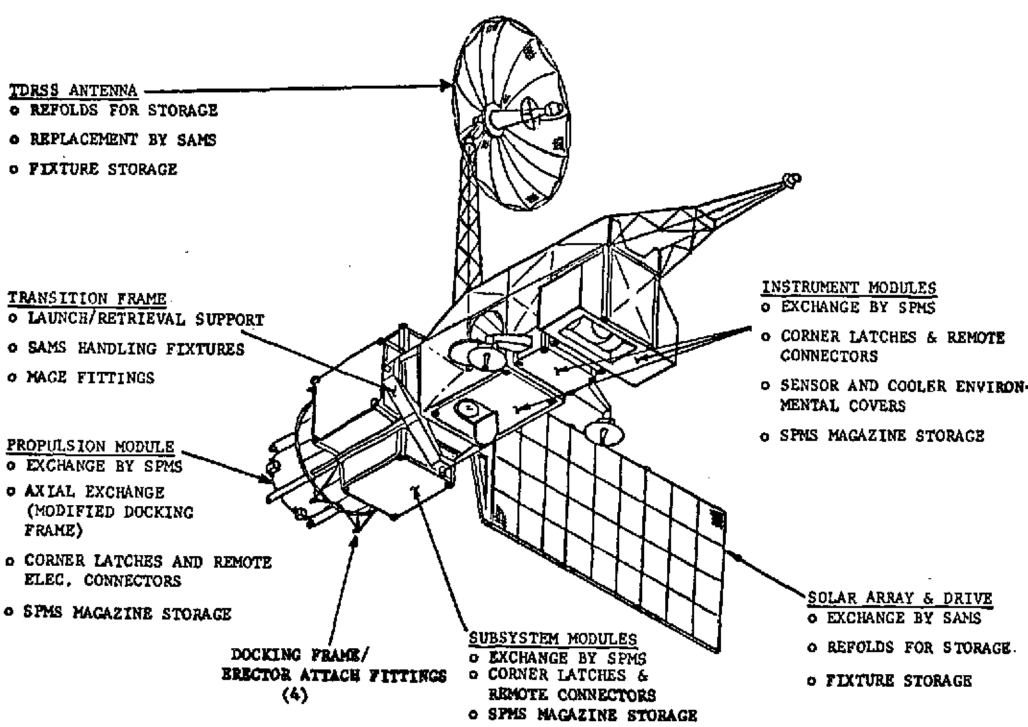
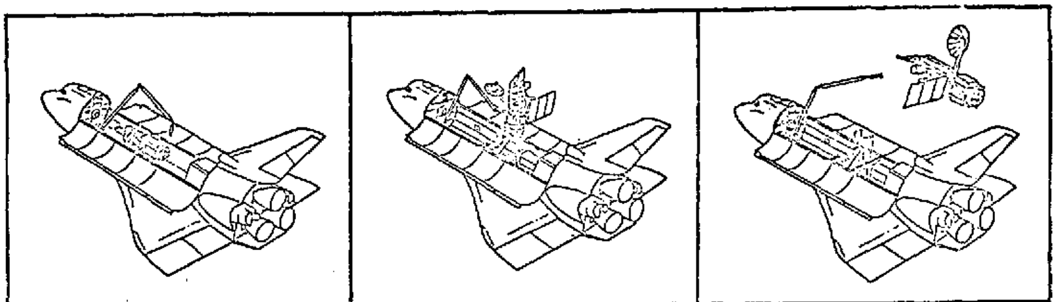


FIGURE 1-3 EOS SPACECRAFT RESUPPLY PROVISIONS

The ELECTRICAL INTERFACES between the EOS and Shuttle Orbiter occur under three basic modes of operation:

1. EOS stowed in the Shuttle bay and attached to the launch/retrieve support
2. EOS attached to the docking frame and erected in a vertical position for module exchange
3. EOS detached from Shuttle with Shuttle in a loiter mode.

The attached modes provide hardwire connections for EOS power and input and output signals: command, telemetry and caution and warning. The detached mode employs RF communications between the spacecraft and the orbiter, with the spacecraft on-orbit and providing its own power. The detached mode is used to provide a check of the entire spacecraft while the Orbiter is on-station. The electrical interfaces for the three modes are summarized in Figure 1-4.



INTERFACE	SPACECRAFT IN SHUTTLE RETENTION CRADLE	SPACECRAFT ATTACHED TO POSITIONING PLATFORM	SPACECRAFT IN SHUTTLE LOITER MODE
POWER	HARDWIRE	HARDWIRE	NONE, INTERNAL S/C
COMMAND	HARDWIRE	HARDWIRE	RF (TO SHUTTLE, GRD OR TDRS)
TELEMETRY	HARDWIRE	HARDWIRE	RF (TO SHUTTLE, GRD OR TDRS)
DATA	HARDWIRE	HARDWIRE	RF (TO GRD OR TDRS)
CAUTION & WARNING	HARDWIRE	HARDWIRE	NONE

FIGURE 1-4 SUMMARY OF ELECTRICAL INTERFACES

All hardwired electrical interfaces between EOS and the Shuttle will be provided through the spacecraft umbilical connector. Cable connections from the umbilical will be made to the power interface panel (Station 695) and to the cabin signal and control interface panel (Station 576) shown in Figure 1-5.

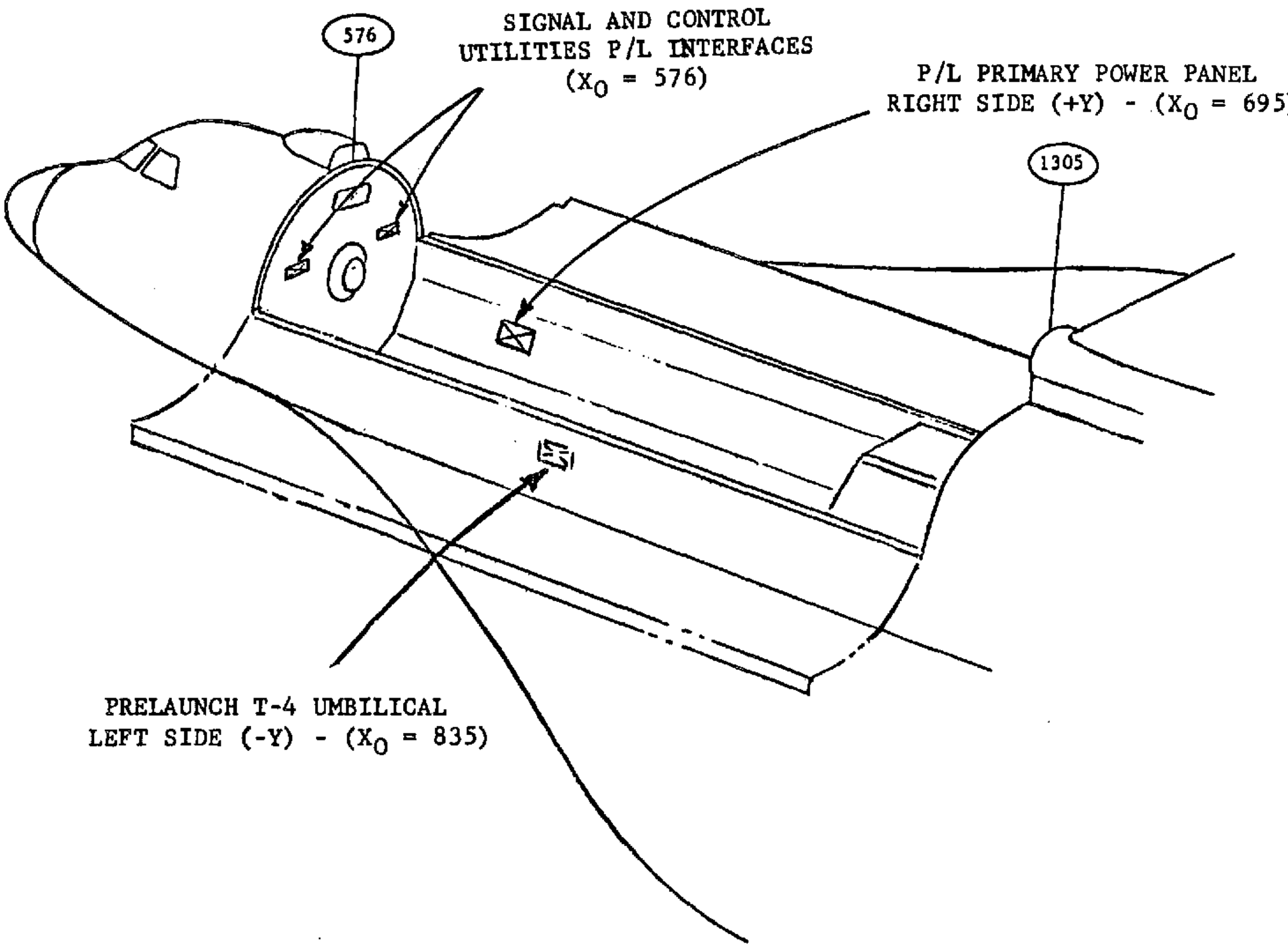
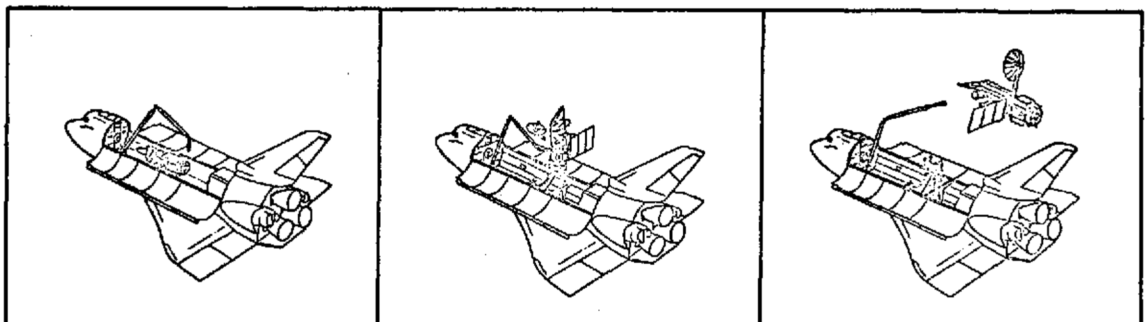


FIGURE 1-5 PAYLOAD/ORBITER ELECTRICAL I/F

The SPACE SHUTTLE has generally been found to be effective in supporting the OPERATIONS of EOS during all phases of the mission. However, in some cases, notably during prelaunch operations, the preferred mode of operation requires modification to fit the overriding Shuttle operational flow. The final close-out of the payload at approximately L-69 is critical since it restricts access to the spacecraft for almost three days prior to launch. Key operational advantages inherent in Shuttle utilization are the added capability of on-orbit checkout which is summarized in Figure 1-6 in addition to the capability of retrieving the spacecraft for ground or on-orbit resupply.



SPACECRAFT IN SHUTTLE RETENTION CRADLE	SPACECRAFT ATTACHED TO POSITIONING PLATFORM	SPACECRAFT IN SHUTTLE LOITER MODE
• CAUTION & WARNING MONITORING • STATUS/LIMIT CHECKING OF SUBSYSTEMS & INSTRUMENTS • SPACECRAFT OBP MEMORY UPDATING • PRE-DEPLOYMENT CHECKOUT - HARDWIRE & MECHANICAL INTERFACES - ELECTRICAL CONTINUITY - VISUAL INSPECTION	• CAUTION AND WARNING MONITORING • DEPLOYMENT OF APPENDAGES • STATUS/LIMIT CHECKING AND PRELIMINARY FUNCTIONAL CHECKING OF SUBSYSTEMS AND INSTRUMENTS • SPACECRAFT OBP MEMORY UPDATING • PRE-SEPARATION CHECKOUT - R.F. INTERFACE - ELECTRICAL CONTINUITY - VISUAL INSPECTION • VERIFY RECAPTURE & RETRIEVAL CAPABILITY	• EOS ON SPACECRAFT ACS, POWER AND COMMUNICATIONS • ACTIVATION AND CHECKOUT UNDER GROUND CONTROL • REMAINDER OF SPACECRAFT DEPLOYMENTS (IF REQ'D)

FIGURE 1-6 SPACECRAFT CHECKOUT IN SHUTTLE ORBIT

The COST BENEFITS accrued from Shuttle utilization is the key study issue addressed in this report. The cost benefits depend, to a great extent, on how Shuttle is used, be it solely as a low cost launch vehicle or as an integral part of the total EOS system accomplishing on-orbit resupply to extend the spacecraft life in orbit. Many variables are involved in evaluating the impacts of alternate Shuttle uses on the EOS program costs. The approach taken was to assume reasonable values for most variables and to also investigate some key variables (such as refurbishment costs, launch cost and number of spacecraft failures) parametrically. These parametric analyses allowed greater insight regarding those variables which most impacted the analysis results. Once this insight was gained, program costs were determined for nominal (best estimate) and maximum (worst case) values of these critical variables. The total program costs were determined for a nominal mission model having two spacecraft in orbit at one time over a ten year lifetime for expendable spacecraft, ground serviceable and on-orbit serviceable spacecraft cases. The results of this analysis are summarized in Table 1-1. In all cases the on-orbit serviced spacecraft (10 year lifetime) proved lowest cost and the expendable spacecraft proved highest cost. The sole difference between Options #1 and #2 under the nominal cost case was the method of charging for spacecraft costs. In Option #1 total spacecraft costs for the number of spacecraft required to perform the mission model were charged (for a 10 year period) independent of the program lifetime that might be expected from these spacecraft. In Option #2 the spacecraft costs were prorated for a ten year period of a longer program. (For example, if three spacecraft are required to perform the mission model but these spacecraft would last 15 years with refurbishment the prorated cost for a ten year program would be only two spacecraft). For each of the nominal cases considered, there is very little difference in cost between the ground serviced spacecraft and the combined ground and on-orbit serviced spacecraft. When the "worst case" variables are considered the combined ground and on-orbit serviced spacecraft show a decided advantage over the ground serviced spacecraft.

TABLE 1-1

SHUTTLE MODE COST ANALYSIS SUMMARY

CASE	NOMINAL COST M$ — OPTION #1 (•NOM. REFURB. •2 FAILURES •TOTAL S/C COSTS •N.R. SERVICE COSTS)	NORMALIZED COST	NOMINAL COST M$ — OPTION #2 (•NOM REFURB. •2 FAILURES •PRORATED S/C COSTS •N.R. SERVICE COSTS)	NORMALIZED COST	ALTERNATE COST M$ (MAX VARIABLES) (•HIGH REFURB. •3 FAILURES •TOTAL S/C COSTS •MOD. HIGH LAUNCH COSTS •N.R. SERVICE COSTS)	NORMALIZED COST
EXPENDABLE SPACECRAFT (SINGLE LAUNCH)	382	1.90	382	1.90	459	1.71
EXPENDABLE SPACECRAFT (DUAL LAUNCH)	382	1.90	382	1.90	428	1.59
GROUND SERVICE SPACECRAFT (SINGLE LAUNCH)	236	1.17	212	1.05	381	1.42
GROUND SERVICE SPACECRAFT (DUAL LAUNCH)	258	1.28	213	1.06	352	1.31
GROUND & ON-ORBIT SERV S/C (1 SERVICE & RETURN)	236	1.17	215	1.07	306	1.14
GROUND & OR-ORBIT SERV S/C (2 SERVICE & RETURN)	233	1.16	213	1.06	300	1.12
ORBIT SERVICE SPACECRAFT (6 YR LIFE)	257	1.28	237	1.18	317	1.18
ORBIT SERVICE SPACECRAFT (10 YR LIFE)	201	1.00	201	1.00	269	1.00

The following conclusions have been made from the cost analysis:

- The most cost effective use of Shuttle is achieved by using Shuttle to deliver the spacecraft and also assist in servicing the spacecraft to extend its orbital lifetime.
- The Shuttle launched EOS spacecraft should be designed for on-orbit servicing while the spacecraft launched prior to Shuttle availability can be designed for Shuttle retrieval and ground servicing without incurring significant cost penalties over on-orbit servicing.
- As designs of EOS and Shuttle mature, the Shuttle analysis can be refined to establish the most cost effective use of Shuttle and the optimum interval for Shuttle service. This may include combined on-orbit and ground servicing or may be limited to on-orbit servicing of the spacecraft.

The SHUTTLE ORBIT ANALYSIS TRADES indicate that it is cost effective to include orbit transfer capability on-board the EOS spacecraft allowing Shuttle delivery and retrieval at an altitude range between 465 and 610 km (250 and 330 nm) independent of EOS mission altitude (see Figure 1-7). The mission impacts of servicing the spacecraft at a low altitude (compatible with Shuttle high payload capability) and returning to the mission orbit of 775 km (418 nm) have been investigated without uncovering any significant problems. This concept of low altitude servicing at a shuttle "base" encourages multiple spacecraft servicing when the spacecraft are in approximately the same orbital plane. The mission altitude selected also permits direct Shuttle access (at higher cost) in the event of a spacecraft failure which prevents its return to the service altitude.

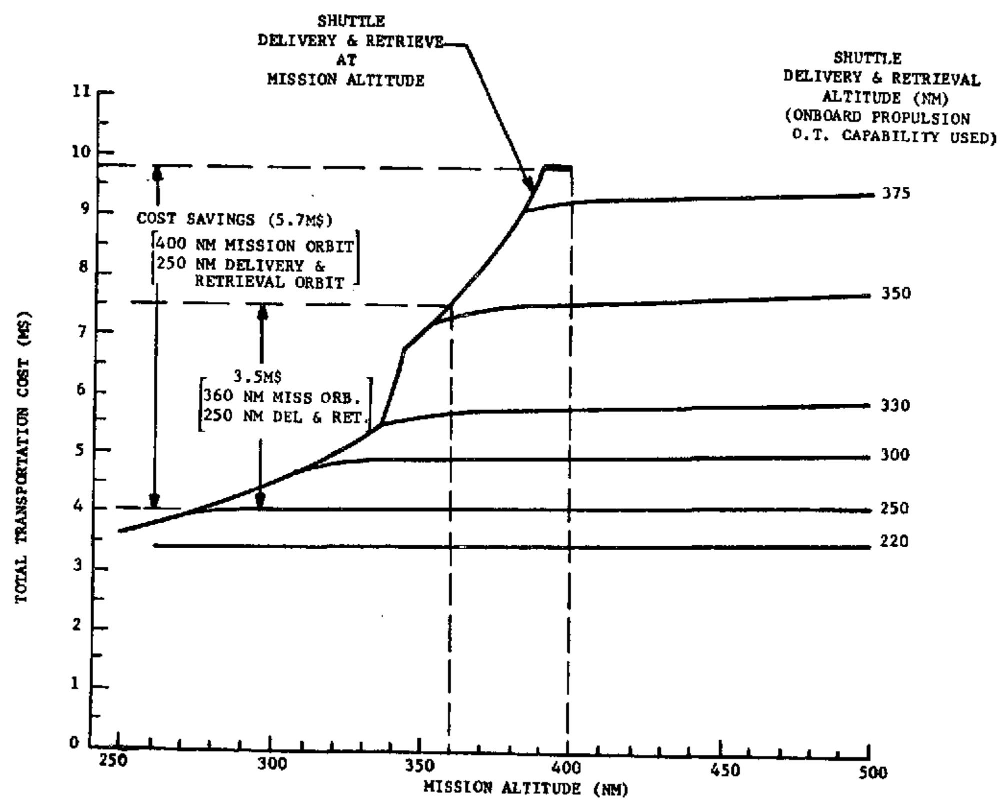


FIGURE 1-7 TRANSPORTATION COST SAVINGS WITH SHUTTLE DELIVERY & RETRIEVAL AT LOW ALTITUDES

The SAFETY CONSIDERATIONS on EOS differ from current automated payload requirements in that these programs are primarily concerned with safety during ground operations while the use of Shuttle requires that this concern be extended into the flight phases of the mission. This added set of requirements potentially impacts all aspects of the system design and development. The results of a safety review indicate however that only relatively minor design modifications are required to "safe" an EOS spacecraft for a Shuttle launch.

A preliminary investigation was conducted of the compatibility of EOS and the Shuttle-induced environment. Two major areas of concern emerged for study. With respect to CONTAMINATION CONTROL it was felt that the control of gaseous and particulate contamination does present a potential problem to the EOS optical instruments. In order to provide for the case where such a condition does exist, several design and operational countermeasures are proposed for both the Shuttle and the EOS. The general conclusion of our analysis is that methods and techniques do exist which can successfully cope with potential contamination problems.

The analysis of Shuttle THERMAL CONTROL provisions was conducted with the currently available Shuttle cargo bay thermal environment data. The results indicate that the spacecraft can be integrated with the Shuttle by taking the following steps:

1. Limit the maximum ground conditioning air temperature to $86^{\circ}F$ to ensure that the battery temperatures are maintained below their maximum allowable transient temperature of $95^{\circ}F$ during launch.
2. Select spacecraft orientations in the Shuttle bay to ensure that critical components (such as batteries) are located away from local "hot spots" that occur in the payload bay during reentry.

SECTION 2.0

STRUCTURAL MECHANICAL INTERFACES

2.1 INTRODUCTION

2.1.1 SHUTTLE EQUIPMENT

Mechanical interface definitions for Shuttle have been taken from RI document No. SD 74-SA-0057, dated June 1974, "Flight Support System for Earth Observation Satellites"; and for the Special Purpose Manipulator System (SPMS) from SPAR/DSMA Document No. SPAR-R.592, dated January 1974, "Design Definition Studies of Special Purpose Manipulator System for Earth Observatory Satellite".

Shuttle Flight Support System (FSS) and SPMS equipment as installed in the Space Shuttle is shown on Figure 2.1. These items are required to provide launch and retrieval support and for on-orbit refurbishment of the EOS spacecraft. The assemblies shown are:

(1) Launch/Retrieval Support Cradle - Supports the spacecraft within Shuttle during launch and retrieval.

(2) Docking Frame and Erection Mechanism (or Positioning Platform) - Erects spacecraft to vertical position for release and positions spacecraft for resupply operations.

(3) SPMS Module Exchange Mechanism - Removes and replaces spacecraft subsystem and instrument modules.

(4) SPMS Module Storage Magazine - Rotating magazine to house replacement and return modules.

(5) Storage Fixture(s) - Required to support large unique spacecraft assemblies such as the solar array and deployable antennas not accommodated in the Module Magazine.

(6) SAMS - Shuttle Attached Manipulator (or Remote Manipulator System) - Articulated, remotely controlled arm with special end effectors used for spacecraft capture and handling during deployment and retrieval, and to exchange large spacecraft assemblies not handled by the SPMS.

SPAR/DSMA is responsible for SPMS definition and all other systems are RI responsibility, as are the overall Shuttle interfaces.

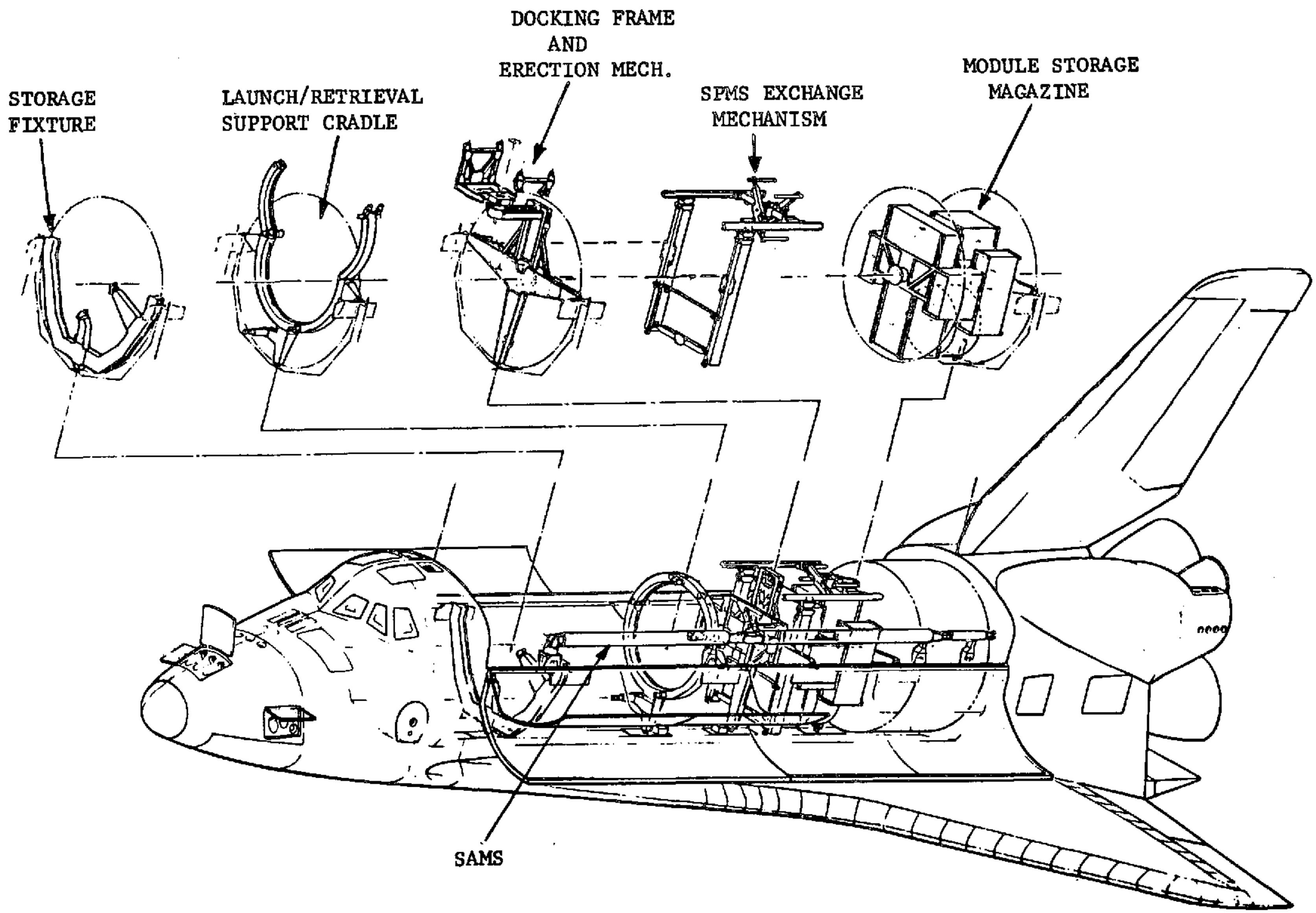


FIGURE 2-1. SHUTTLE EOS FLIGHT SUPPORT SYSTEM

2.1.2 REFERENCE EOS SPACECRAFT DESIGNS

Two types of EOS spacecraft design have evolved for Shuttle era applications. The Thematic Mapper and HRPI instruments have been used for both designs as representative payloads used to evaluate Spacecraft/Shuttle compatibility and interfaces. The first configuration, Figure 2-2, has been designed for initial launch by the Delta booster (or Shuttle) and retrieval by Shuttle. As shown in the exploded view the spacecraft is of modular design with separate subsystem, propulsion, and wideband modules. These modules are rigidly attached to the spacecraft structure, as are the individual instruments, and are not resuppliable by Shuttle. Large deployable appendages such as the solar array and TDRSS antenna refold for retrieval. This retrievable spacecraft is representative of the maximum weight and volume capability of the Delta L/V and would be capable of accepting a wide variety of alternate payloads for retrieve-only missions. The Thematic Mapper plus dual MSS has also been used during the evaluation as an alternate payload.

The second configuration, Figure 2-3, is also a modular arrangement designed for initial launch on Titan (or Shuttle) with Shuttle retrieve and on orbit resupply capability. Subsystem modules are identical to the retrievable-only design except for the addition of remote latches and electrical disconnects for resupply. Instruments are separately mounted in replaceable modules, and the propulsion module has been designed for exchange using the SPMS. Special provisions are required for exchange of the solar array and TDRSS antenna assemblies using the SAMS manipulator. This spacecraft is 400 to 500 pounds heavier than the retrieve-only design and is larger to accommodate replaceable modules, requiring additional fairing volume over the retrievable design. These additional weight and volume requirements have necessitated selection of Titan rather than Delta for a conventional booster initial launch.

These two spacecraft have been designed to meet defined Shuttle and SPMS interface requirements wherever possible, and proposed changes in Shuttle equipment design and/ or interfaces have been identified.

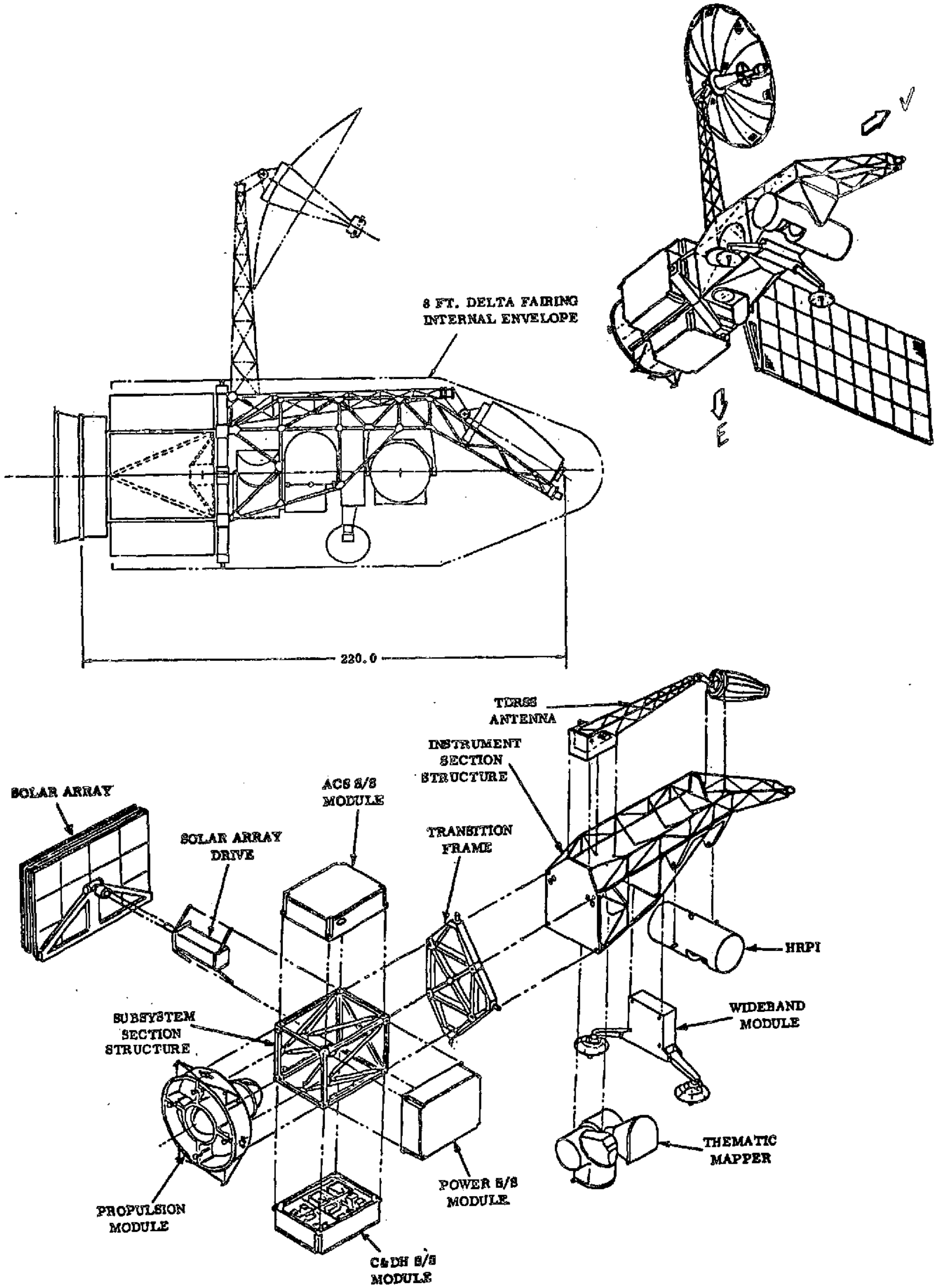


FIGURE 2-2. EOS SPACECRAFT RETRIEVABLE CONFIGURATION

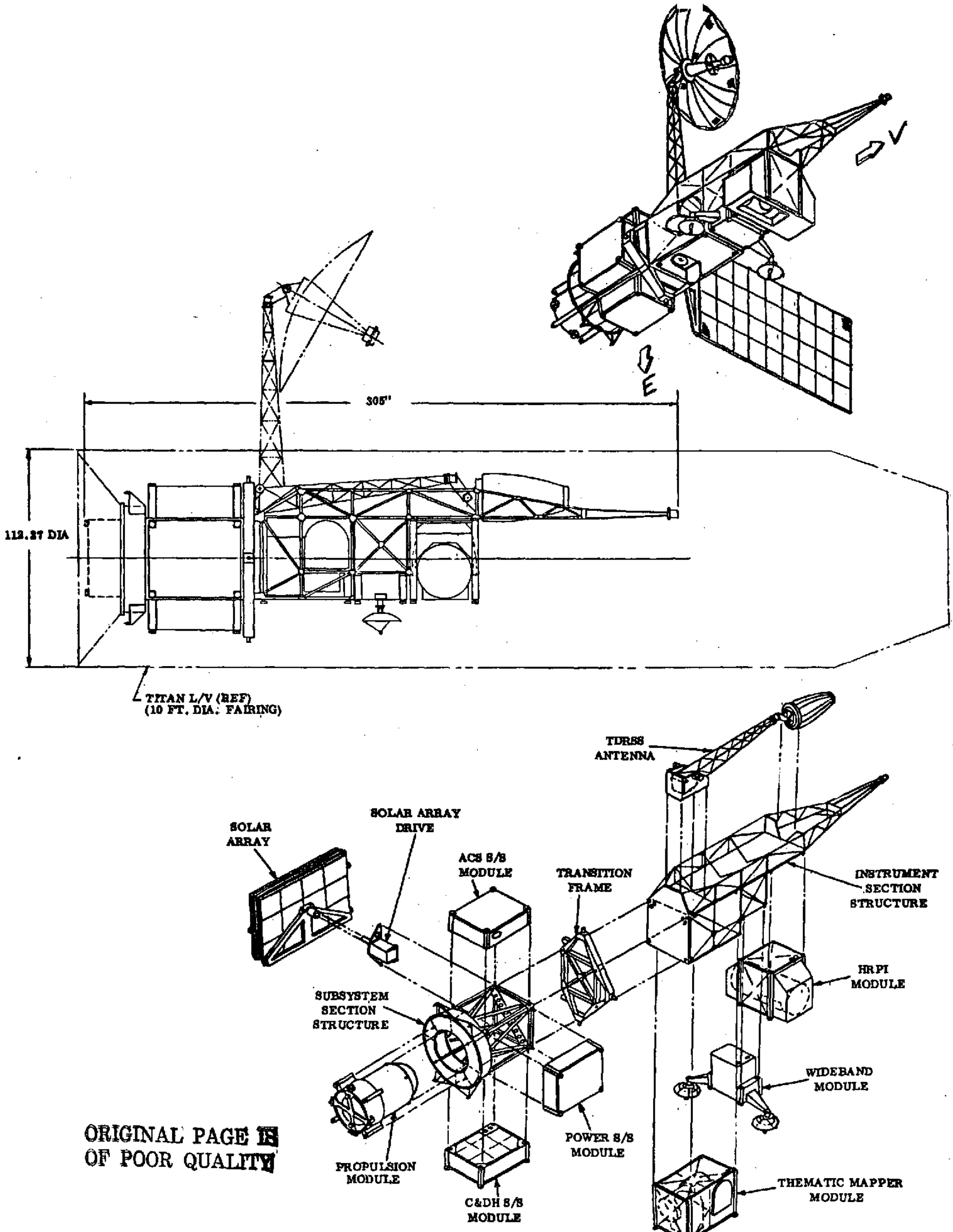


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FIGURE 2-3. EOS SPACECRAFT-RESUPPLY CONFIGURATION

2.2 SHUTTLE INTERFACES AND STRUCTURAL CRITERIA

2.2.1 CARGE BAY ENVELOPE

Shuttle cargo bay available volume and the location of payload retention fittings are shown on Figure 2-4. The EOS payload envelope is 180 inches in diameter and 623 inches in length allowing 97 inches in length for installation of two OMS kits in the bay aft section.

Payload retention fittings are spaced basically on a 59-inch pitch along the bay length. Currently defined locations of FSS and SPMS supports are shown on Figure 2-4.

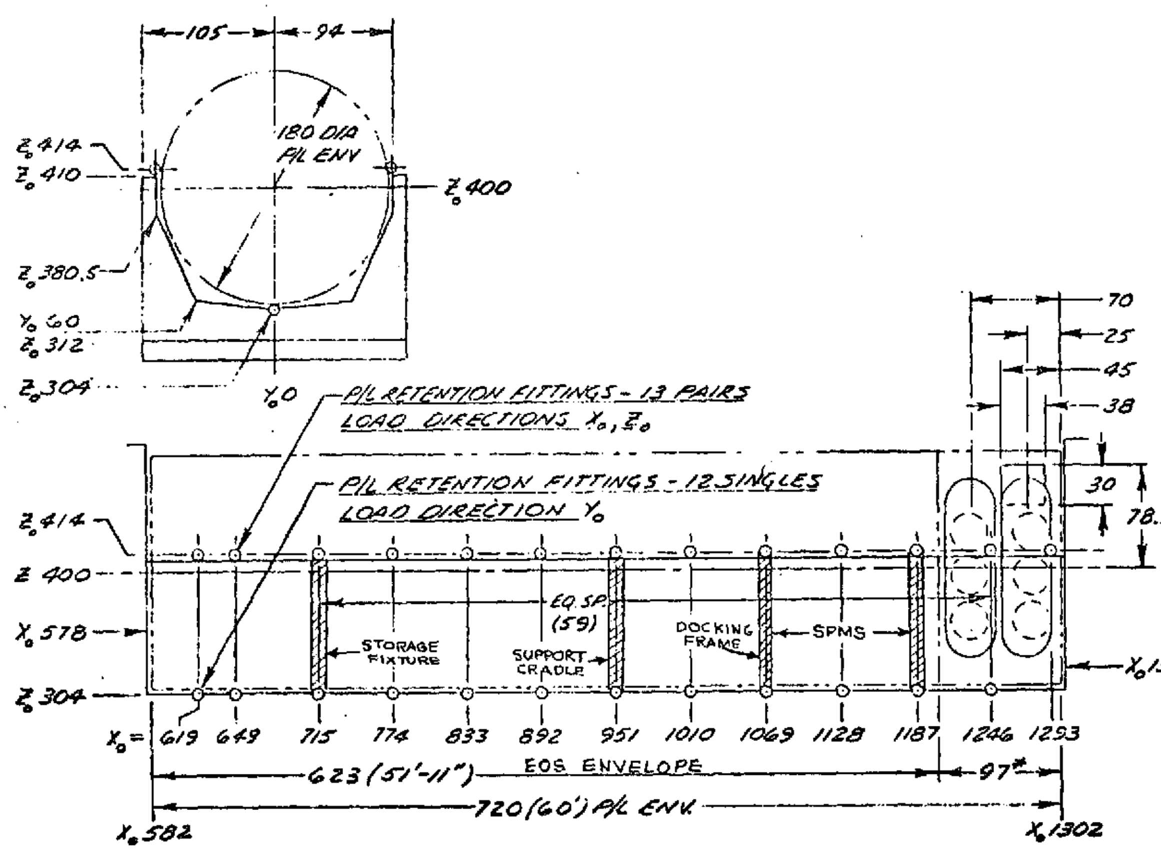
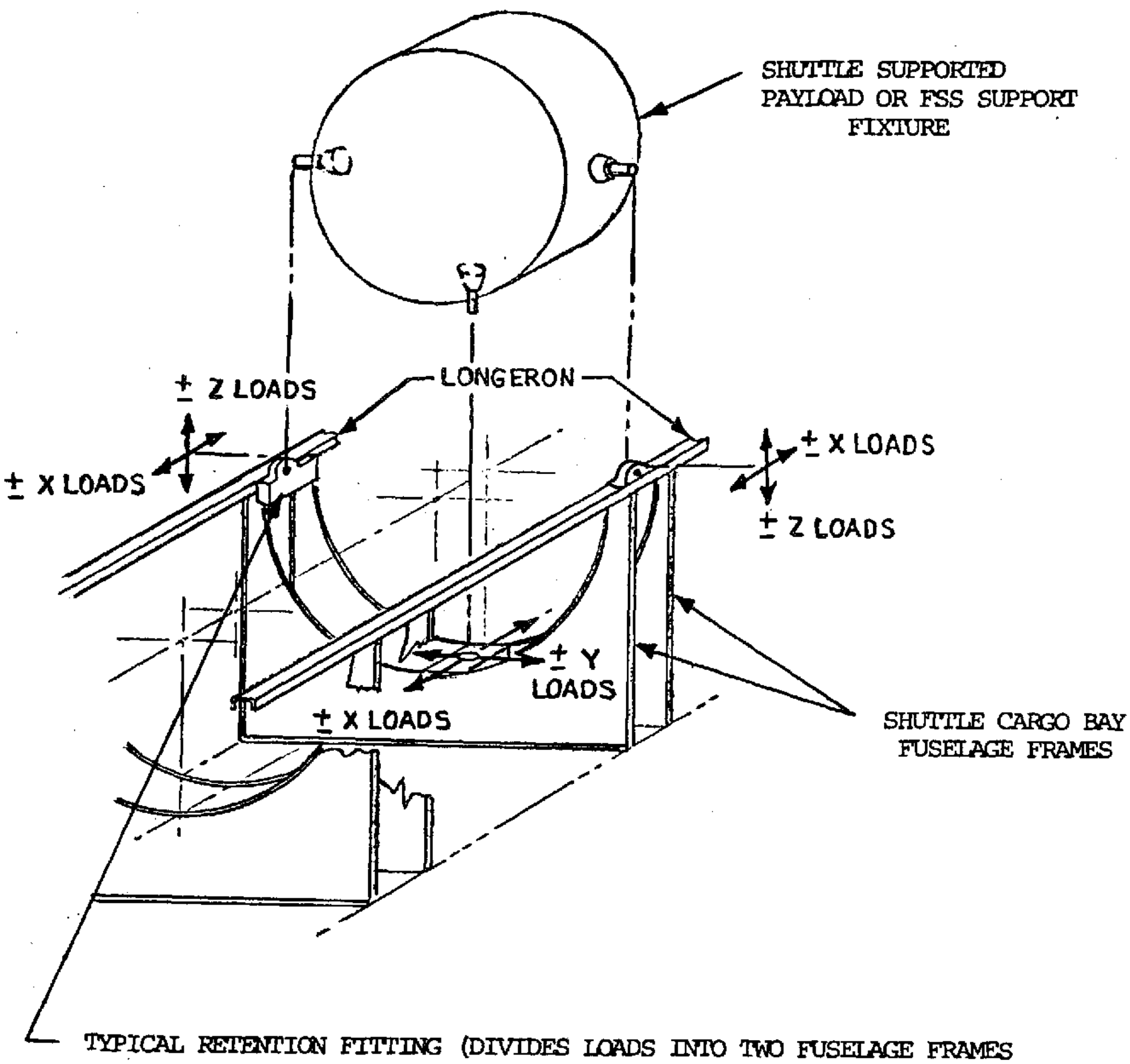


FIGURE 2-4 SHUTTLE CARGO BAY ENVELOPE

2.2.2 SUPPORT REACTION SYSTEM

The originally recommended support system for Shuttle bay payload attachment used a statically determinant four-point arrangement which reacted only side (Y) loads at the lower fitting requiring forward or aft pitch links to react turning loads about the longeron fittings. This system has been replaced by RI for the FSS assemblies with the three-point reaction system shown on Figure 2-5, where the turning moment is reacted by $\pm$X loads at the lower centerline fitting. This system simplifies the FSS design in eliminating the pitch links, and as with the previous four-point system, minimizes induced loads into the supported payload due to Shuttle body deformations. Either of these reaction systems will result in equivalent loadings on the spacecraft and the three-point system has been used as the EOS baseline.



REF: RI DOCUMENT #SD74-SA-0057, DATED JUNE 1974
FIGURE 2-5. SPACE SHUTTLE THREE POINT PAYLOAD RETENTION SYSTEM

2.2.3 STRUCTURAL DESIGN CRITERIA

Shuttle loads and dynamic environments are summarized in Table 2-1 along with Delta and Titan IIIB criteria. Shuttle induced loads on the spacecraft are reacted by the FSS cradle at the transition ring or frame between the spacecraft subsystem and instrument sections, while spacecraft body loads from the instrument section are carried through

TABLE 2-1 STRUCTURAL DESIGN CRITERIA

LAUNCH SYSTEM	SPACECRAFT QUALIFICATION TEST LEVELS (1.5 X EXPECTED LEVEL)							S/C LOAD FACTOR	S/C ULTIMATE DESIGN LOADS (G'S)	
	ACCELERATION (G'S)		RANDOM VIB. (G RMS)	MAX. SINE VIB. (G'S)		ACOUSTICS db	SHOCK RESP. (G'S MAX.)			
	THRUST	LATERAL		THRUST	LATERAL				THRUST	LATERAL
Delta	-18.0	±3.0	14.1	6.0	2.0	144	1700	2.0	-36.0	6.0
Titan IIIB	-13.5	±2.5	16.9	3.0	2.0	145	3900	2.0	-27.0	5.0
Shuttle L/O B/O Entry Ldg Crash	-3.45 -4.95 ± .38 ±2.25 +9.0	1.28 .81 4.56 3.8 4.5	7.9 to 24.3	TBD	TBD	143 to 149	TBD	2.0 (1.2 crash)	- 6.9 - 9.9 ± .76 ±4.5 +10.8	2.56 1.61 9.12 7.6 5.4

the subsystem section structure to the conventional adapter on Delta or Titan. This central body support provided by the transition frame reduces loads in the subsystem section for Shuttle retention, therefore the Delta and Titan acceleration levels will produce higher loadings in the subsystem section than those experienced in shuttle. The Instrument Section lateral acceleration loads are slightly higher for the Shuttle landing condition; however, the combined thrust and lateral conditions for Delta or Titan produce higher overall loads in this section. The most potentially severe structural loadings from Shuttle appear to be the random vibration and acoustic noise levels which may produce the highest dynamic response in the Instrument Section and govern instrument mounting and equipment design.

2.2.4 PRESSURIZED COMPARTMENT INTERFACES

The man-machine interface for Shuttle deployment, retrieval and resupply operations is performed using control panels located in the forward pressurized crew compartment. These control panels are located in the Payload Specialist Station (PSS) in the aft position of the cabin. Spacecraft control or monitoring equipment in standard racks will also be installed in this area; however, at this time the equipment size, location, and detail functions have not been defined.

A view port on the aft cabin bulkhead is provided along with remote television cameras to permit direct viewing of the cargo bay. All normal Shuttle operations are by remote control with no EVA currently planned.

2.2.5 SHUTTLE ATTACHED MANIPULATOR SYSTEM (SAMS)

The SAMS or Remote Manipulator System (RMS) is used to deploy the spacecraft from the FSS and to capture and remate the spacecraft to the docking platform for servicing or retrieval. The SAMS is also used to remove and replace spacecraft appendages not accomplished by the SPMS. This device is under development and the geometry and characteristics summarized represent the current status from RI data.

SAMS geometry and joint articulation is shown on Figure 2-6. The total length is 50 feet and boom diameter is 1-1/4 feet. The single SAMS normally provided is stowed on the left side of the orbiter and is capable of removing a 15-foot diameter, 60-foot long, 65,000-pound payload without exceeding 3 inches side and end clearance. The boom contains lights and TV cameras for payload viewing during operations, and a second system can be installed on the right side of the orbiter for a 730-pound weight penalty chargeable to the payload.

Specific end effectors for SAMS have not been defined, at this time,for the proposed EOS applications.

The manipulator operating envelope is given in RI Report SD-74-SA-0057 and the maximum torque performance of each flexible joint is summarized in Table 2-2.

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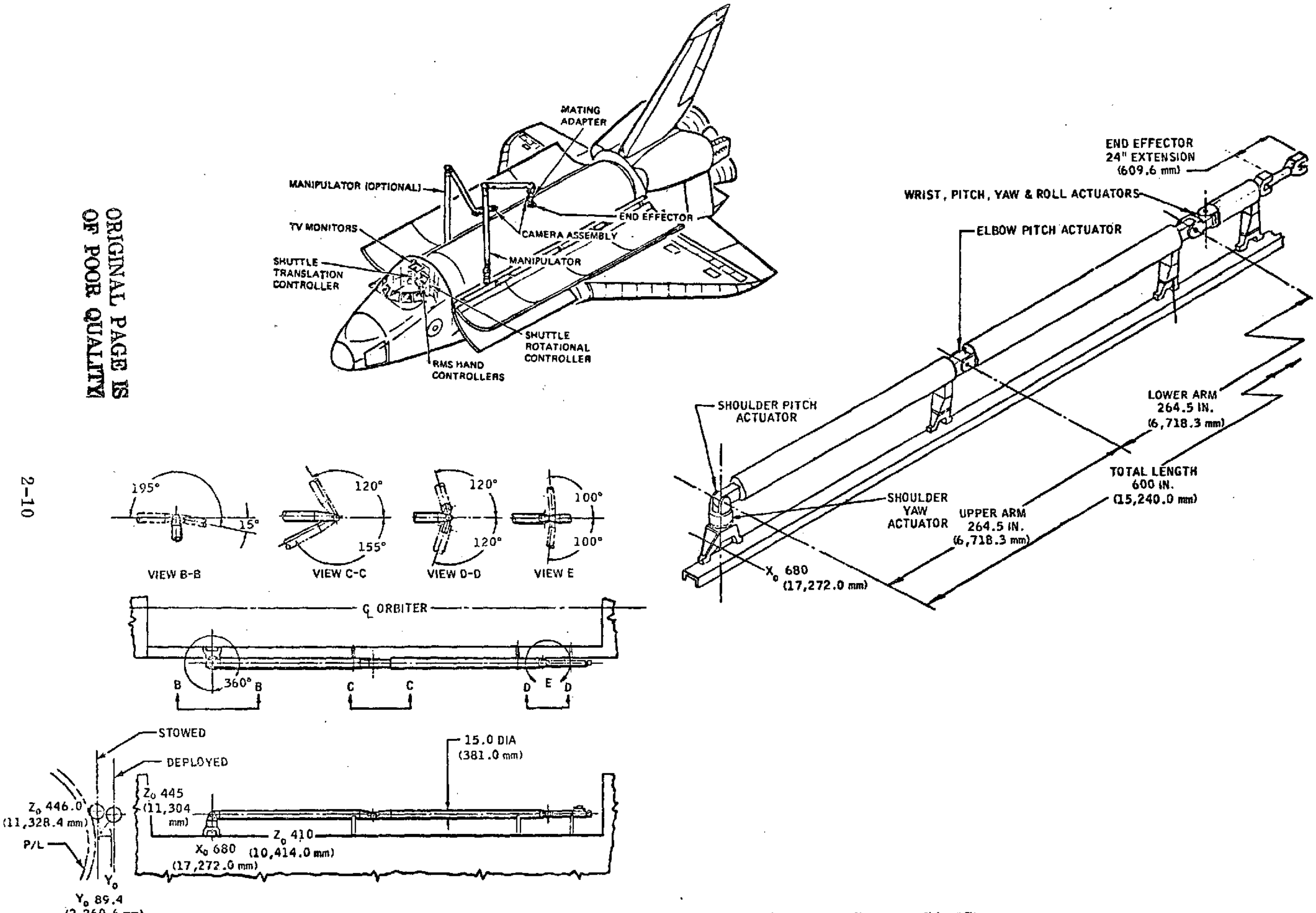

FIGURE 2-6. SHUTTLE ATTACHED MANIPULATOR SYSTEM (SAMS)

Table 2-2 SAMS Maximum Joint Performance (Ref. Figure 2-6)

Joint			Torque
Shoulder	(View B-B)	- pitch	6000 in-lb
		- yaw	6000 in-lb
Elbow	(View C-C)	- pitch	3600 in-lb
		-roll	2400 in-lb
Wrist	(View D-D)	- pitch	2400 in-lb
		- yaw	2400 in-lb

Allowable payload (spacecraft) dynamics prior to retrieval by SAMS are:

1. Maximum Limit Cycle (Inertial) =	$\pm$ 1 deg about any axis
2. Maximum Limit Cycle Rates (Inertial) =	$\pm$ 0.1 deg/sec about any axis
3. Allowable Attach Point or Docking Ring Motion (Relative) =	$\pm$ 3.0 in.

The baseline EOS ACS system pointing accuracy of $\pm$ .01 deg and rate of $\pm 10^{-6}$ deg/sec are well within the allowable spacecraft dynamics specified above. The ACS back-up mode pointing accuracy of $\pm$ 6 degrees exceeds the allowable pointing error of $\pm$ 1 degree. The back-up mode rate of $\pm$ 0.5 deg/hr. is well within the requirement of $\pm$ 0.1 deg/sec which is felt to be the more critical requirement. The specified allowable spacecraft dynamics must be resolved since placing a pointing accuracy of $\pm$ 1 deg on a retrievable payload will significantly limit SAMS use in payload retrieval.

2.3 SPACECRAFT RETENTION AND DEPLOYMENT

2.3.1 LAUNCH/RETRIEVAL RETENTION

The EOS spacecraft is supported by the FSS cradle at Shuttle Station 951 during launch or retrieval as illustrated on Figure 2-7. Redesign of the baseline FSS cradle is recommended to support the spacecraft at a three-point Transition Frame interface rather than with circumferential clamp retention of a Transition Ring. The spacecraft three-point Transition Frame located between the Subsystem and instrument sections provides identical support to the forward and aft spacecraft sections during Shuttle

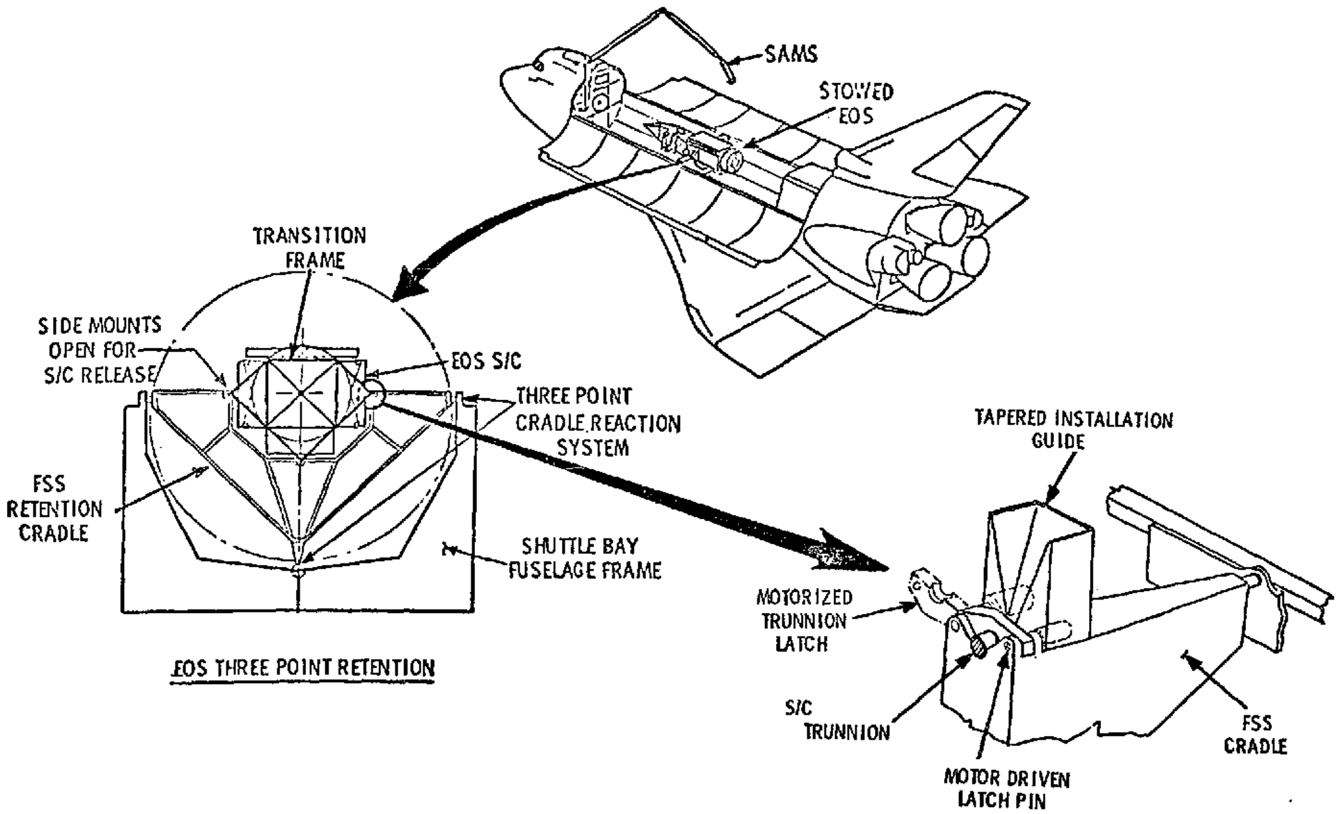

FIGURE 2-7 EOS SHUTTLE LAUNCH/RETRIEVAL RETENTION

retention and reacts loads into the cradle as shown on Figure 2-8. For a Delta or Titan launch the Transition Frame is not used as a load path and body loads are carried through the subsystem section to a conventional aft adapter. The Frame reaction system is identical to the three-point system used for FSS to Shuttle attachment and this design result in significantly lighter cradle and transition structure designs. Each of the spacecraft sections attach to the frame at four corner fittings and special attachments are provided for SAMS handling during deployment and retrieval and for attachment of MAGE fixtures for ground handling and mating of the spacecraft. Cradle attach fittings could be designed for this system to incorporate tapered guides for the three mating fittings providing added tolerances for ease of mating, possibly permitting use of SAMS alone to install the spacecraft in the cradle for retrieval.

The Transition Frame geometry shown on Figure 2-8 has been dictated primarily by the 86-inch diameter Delta shroud envelope.

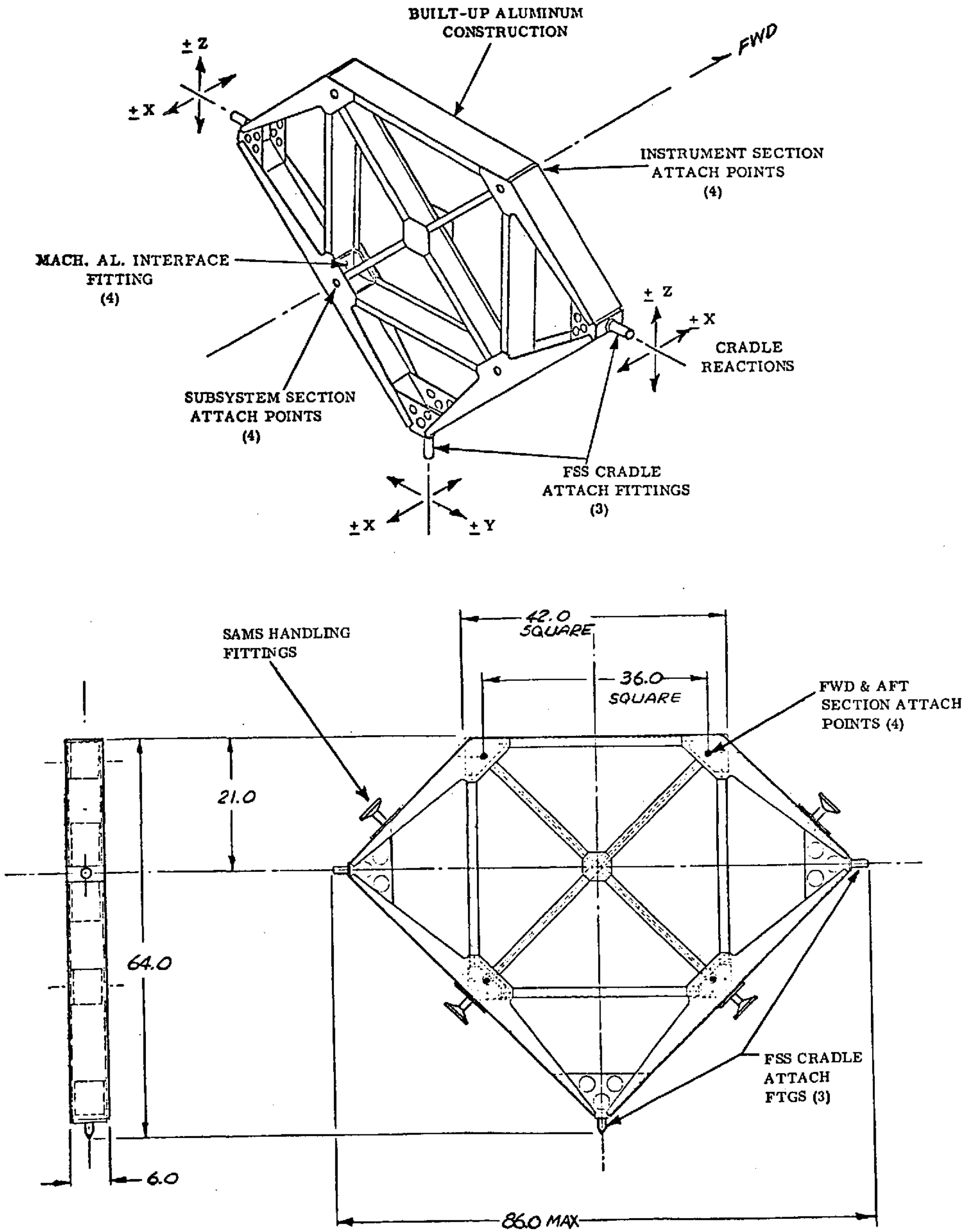

FIGURE 2-8 EOS SPACECRAFT TRANSITION FRAME

2.3.2 SPACECRAFT DEPLOYMENT

The FSS Positioning Platform, Figure 2-9, located aft of the installed spacecraft at Shuttle Station 1069 provides the deployment and docking functions summarized as follows:

1. The docking latches provide the hard dock required to retain and position the EOS while it is extended from the payload bay.
2. The extend and retract movement and the rotation of the EOS about its longitudinal axis to permit access for the remove and replace operations are provided by the 90-degree lift and the rotation mechanisms.
3. The umbilical which connects the EOS to the shuttle control and monitoring circuits will be adjacent to the docking points.
4. After stowage of the EOS in the retention cradle it is necessary to retract the docking latches to decouple the load path and ensure that all loads are carried through the cradle.

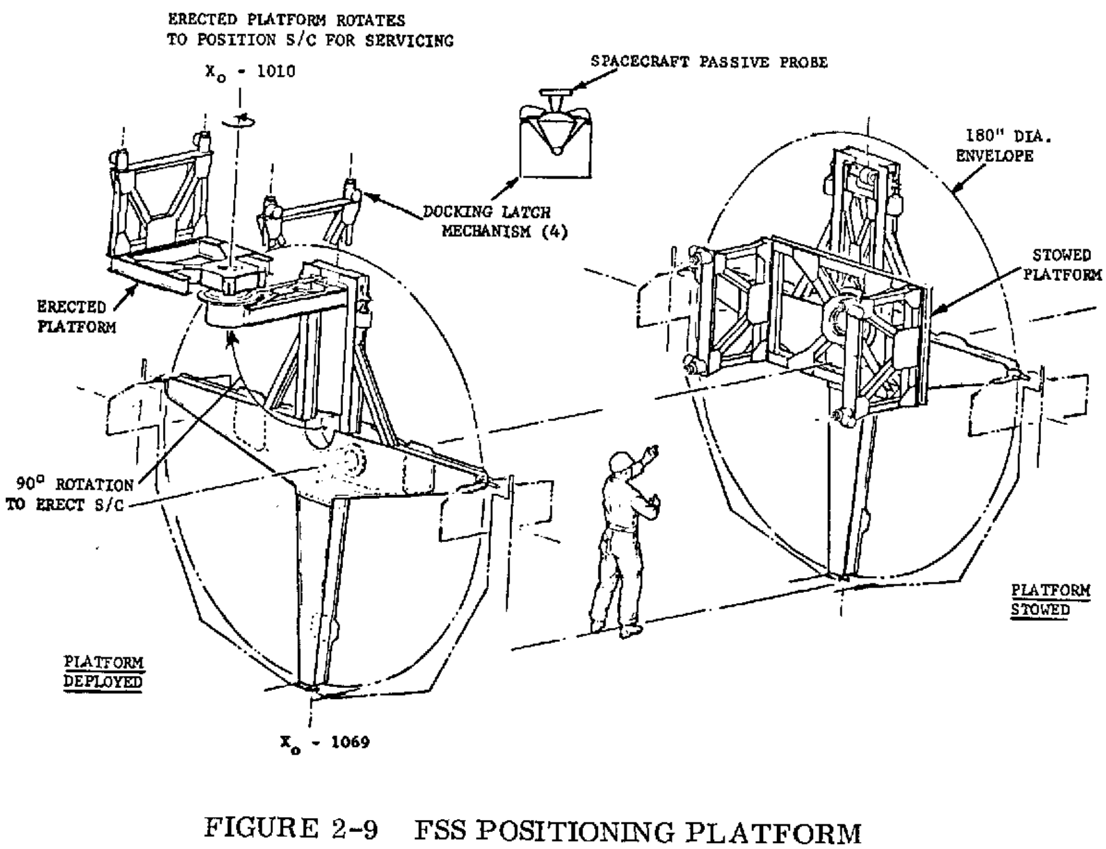

FIGURE 2-9 FSS POSITIONING PLATFORM
(DOCKING FRAME AND ERECTION MECHANISM)

Four active remotely controlled capture latches are provided to engage four passive drogue attachments on the spacecraft. The sizing of the docking mechanism is based on an assumed worst case approach velocity of 2 inches per second, a 10-degree (20-degree cone) approach angle, and 2-inch lateral displacement.

The spacecraft is deployed after orbit insertion by first engaging the docking latches and releasing the spacecraft cradle retention latches. The docking platform next translates the horizontal spacecraft upward one inch to clear the cradle latches, and the spacecraft and platform are rotated 90 degrees to position the spacecraft vertically for deployment or servicing.

The four passive docking interface fittings are shown mounted on the spacecraft on Figure 2-10. The geometry shown is applicable to either the retrieve or resupply configurations and has been reconfigured from the original RI geometry for the smaller EOS configurations designed for Delta or Titan launch.

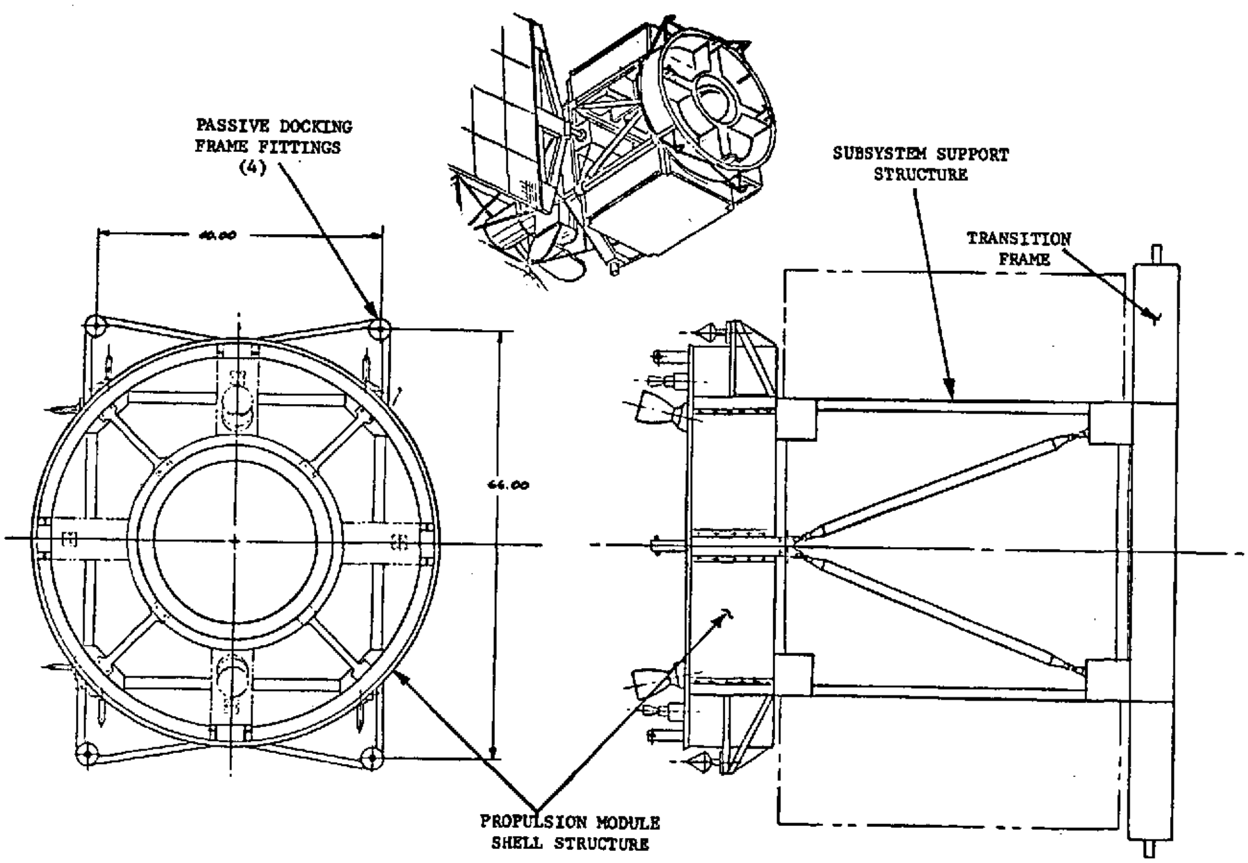

FIGURE 2-10 EOS SPACECRAFT DOCKING ATTACH FITTINGS

2.3.3 SPACECRAFT SHUTTLE INSTALLATION

The spacecraft installed in the Shuttle cargo bay is illustrated on Figure 2-11, showing the EOS mounted horizontally in the Retention Cradle for launch or retrieval and erected by the Positioning Platform for deployment. Figure 2-11 shows a single EOS spacecraft centerline mounted for deployment by the Platform and an alternate arrangement of two spacecraft mounted side by side for deployment by SAMS. This latter, two spacecraft arrangement, shows the design flexibility of the smaller EOS spacecraft design for advanced multiple mission launches and also will provide additional cargo bay space for payload sharing.

The EOS spacecraft will incorporate provisions for MAGE interface fittings on the Transition Frame and aft spacecraft structure to permit either normal horizontal installation into the cargo bay or lateral emergency spacecraft removal with the Shuttle erected vertically on the launch pad.

2.4 SPACECRAFT RETRIEVAL

2.4.1 SHUTTLE RETRIEVAL SEQUENCE

The EOS uses its on-board propulsion capability to return from the operational orbit of 420 nm to the Shuttle rendezvous altitude of 250 to 330 nm. The Shuttle captures and mates the spacecraft to the erected FSS Docking Platform using the SAMS manipulator. The TDRSS antenna and solar array are refolded and secured and sensor and cooler covers closed prior to return of EOS to the cargo bay. The spacecraft is next rotated 90° by the FSS erection mechanism and remated to the Retention Cradle. Once the cradle retention latches are engaged, the docking latches are remotely opened to decouple the spacecraft from the Docking Frame completing stowage for Shuttle retrieval.

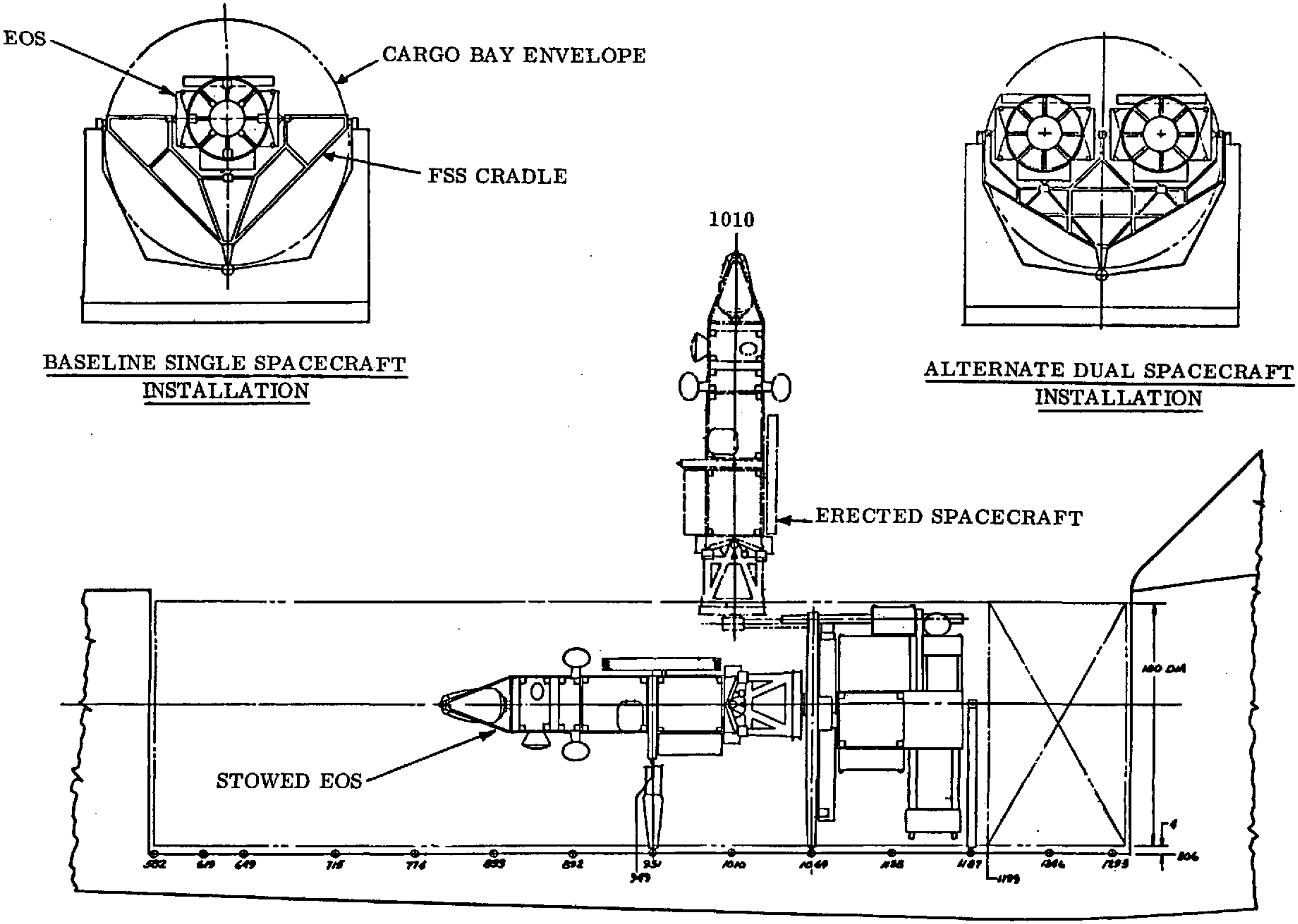

FIGURE 2-11 EOS SHUTTLE INSTALLATION

2.4.2 SPACECRAFT LAUNCH/RETRIEVAL PROVISIONS

EOS spacecraft provisions to accommodate Shuttle launch and retrieval operations and equipment are summarized on Figure 2-12 for the retrievable spacecraft configuration and are also incorporated in the Resupply Spacecraft discussed in Section 2.5.

The features include:

(1) Restowable TDRSS Antenna - Reflector refurls and boom folds using base mounted deployment mechanism. (Emergency jettison by SAMS)

(2) Solar Array refolds and restows on spacecraft as illustrated on Figure 2-13, using SAMS assist. (Emergency jettison by SAMS)

(3) Instrument sensor and coolers close for launch and retrieval.

(4) Transition Frame provides Shuttle launch retention interface and fittings for SAMS and ground support handling equipment.

(5) Docking Frame interface fittings are attached to the aft spacecraft structure.

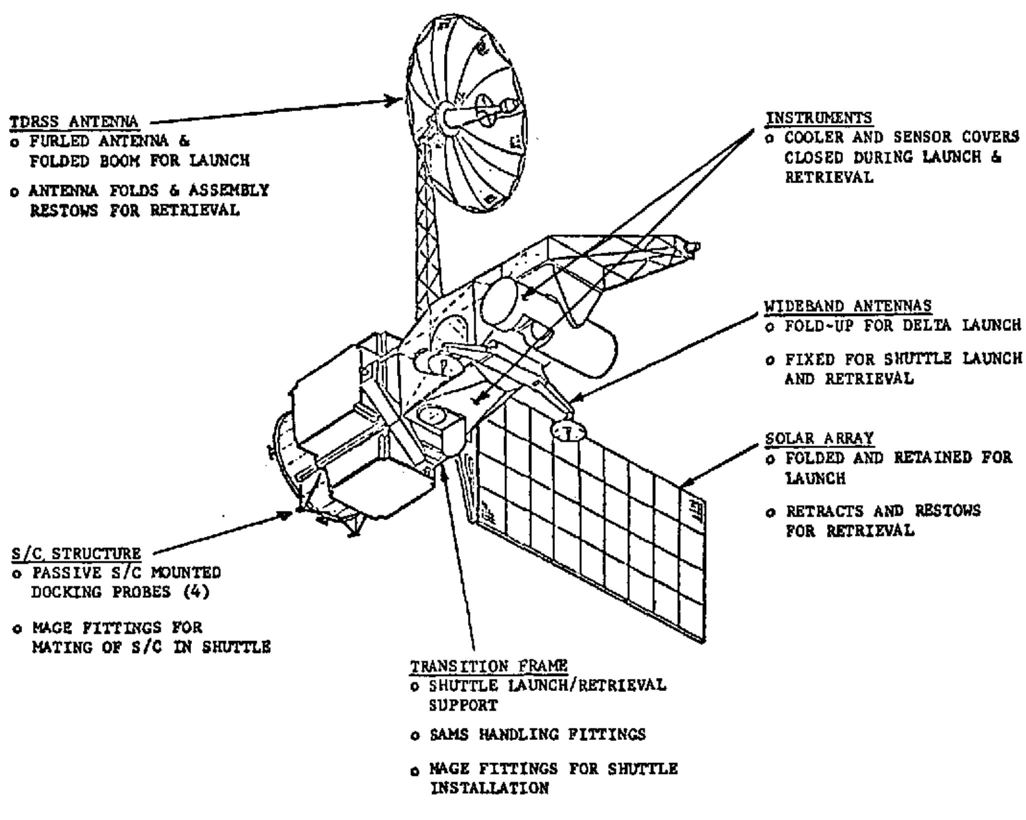

FIGURE 2-12 EOS SPACECRAFT LAUNCH/RETRIEVAL PROVISIONS

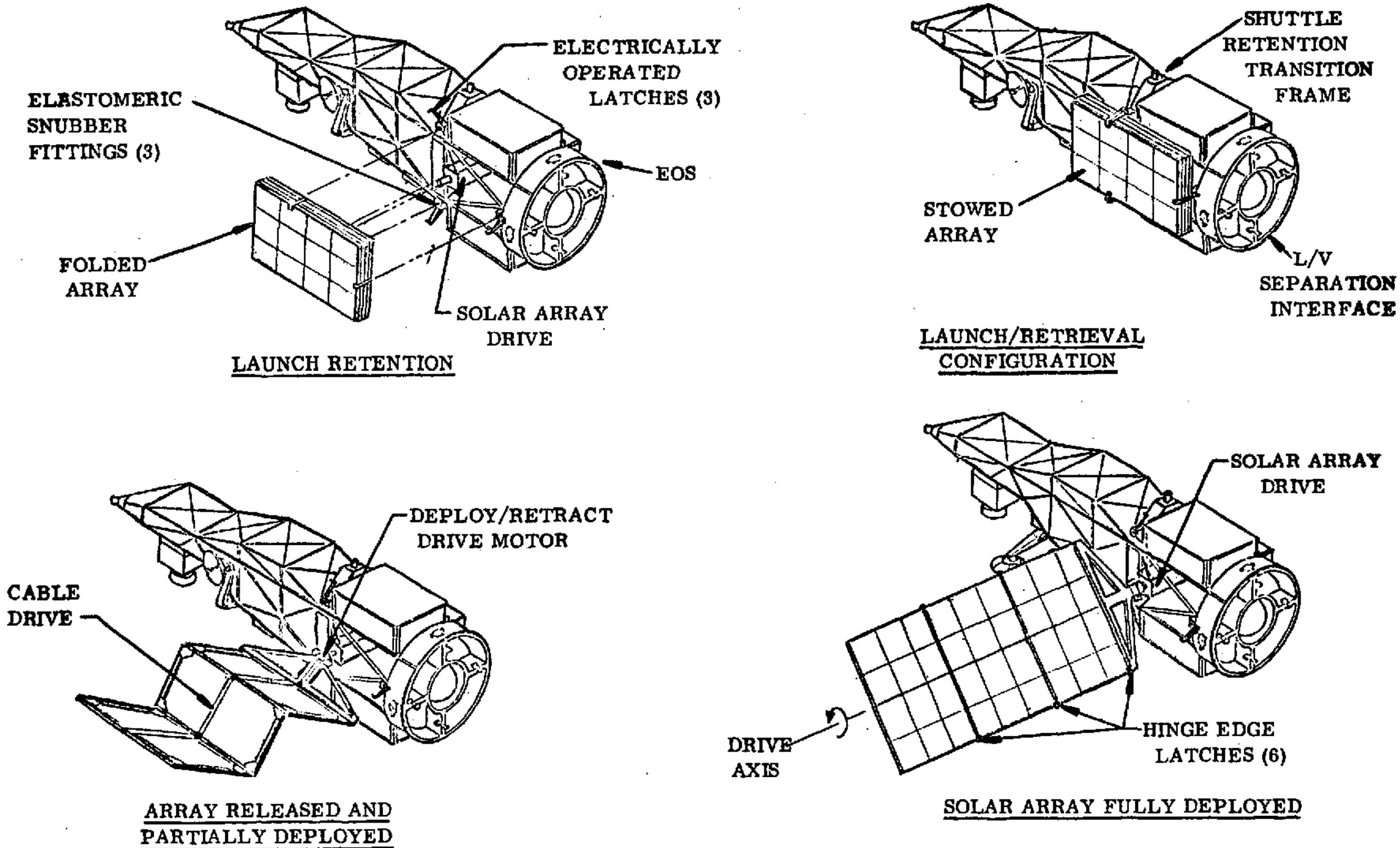

FIGURE 2-13 EOS SPACECRAFT SOLAR ARRAY RETENTION AND DEPLOYMENT

2.5 SPACECRAFT RESUPPLY

2.5.1 MODULE RESUPPLY

The Shuttle EOS resupply mission is illustrated by Figure 2-14 which shows the spacecraft erected on the Docking Platform and positioned for module resupply using the SPMS exchange mechanism. Replacement and returning modules are housed in the SPMS rotating magazine located aft of the exchange mechanism. This concept permits on-orbit replacement of subsystem and instruments by Shuttle resulting in significant program cost savings as discussed in Section 5.0 of this report.

Module resupply as shown on Figure 2-15, requires modular mounting of instruments and subsystems and incorporation of SPMS activated latch mechanisms and remote electrical disconnects.

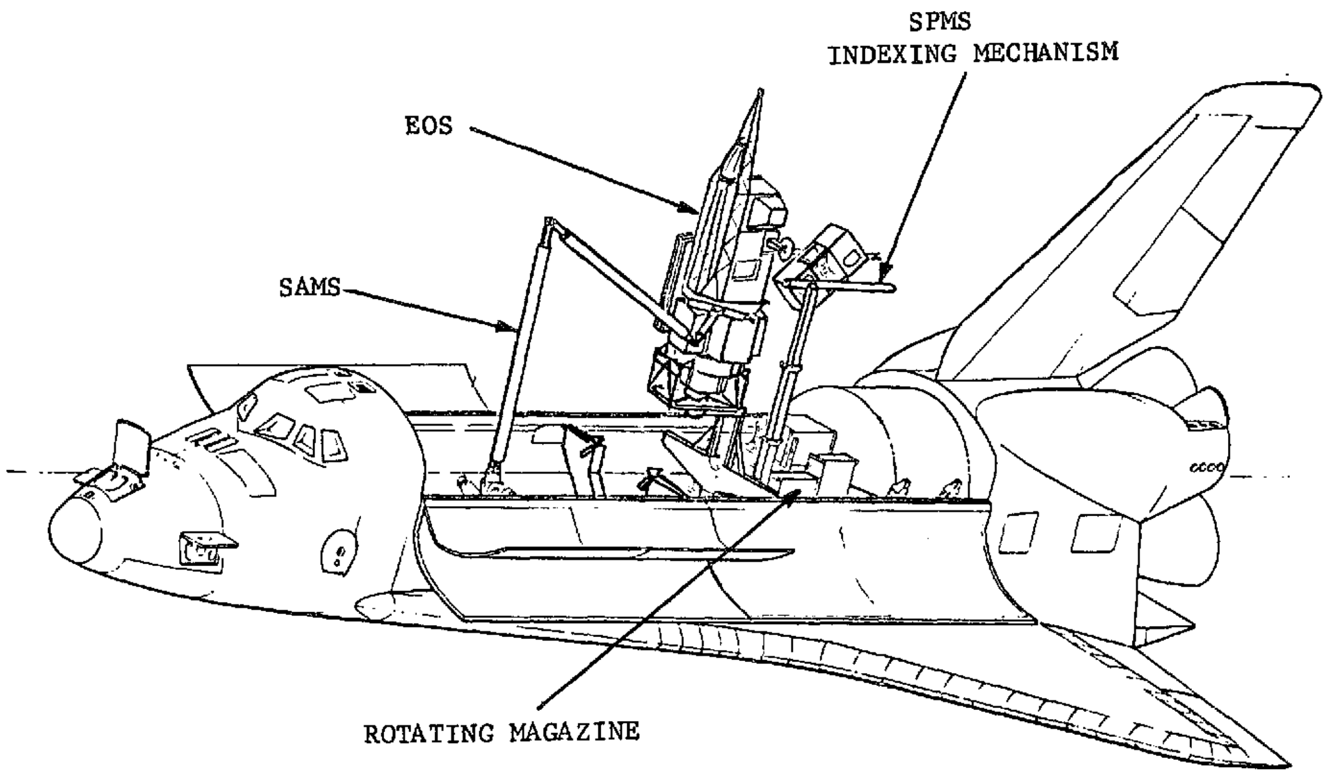

FIGURE 2-14 EOS-SHUTTLE RESUPPLY MISSION

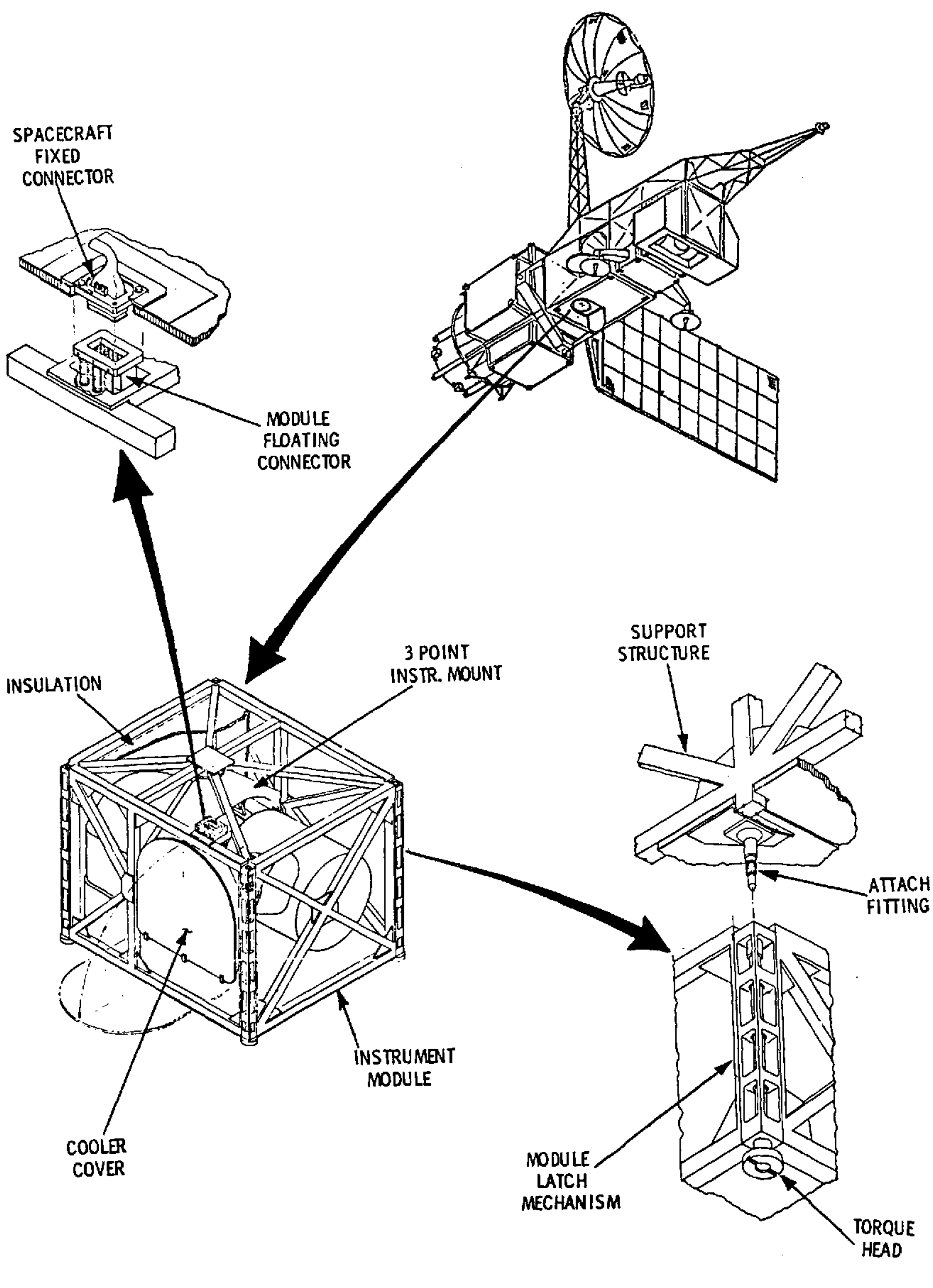

FIGURE 2-15 EOS MODULE RESUPPLY CONCEPT

2.5.2 SHUTTLE RESUPPLY EQUIPMENT

The SPMS installation, as shown on Figure 2-16, consists of two primary assemblies, the Module Exchange (Indexing) Mechanism, and the rotating storage magazine. The items are fully defined in SPAR/DSMA Report SPAR-R.592 dated January 1974. The Module Exchange Mechanism, shown in stowed and deployed configurations on Figure 2-17, has vertically telescoping columns, fore and aft translation rails, and a scissoring Terminal Device to extract and replace modules from the Magazine and spacecraft. This system is used for exchange of the following modules:

(1) ACS, Power, and C&DH Subsystem Modules

(2) Propulsion Module

(3) Wideband Module

(4) Instrument Modules (TM and HRPI on the Reference Resupply Spacecraft)

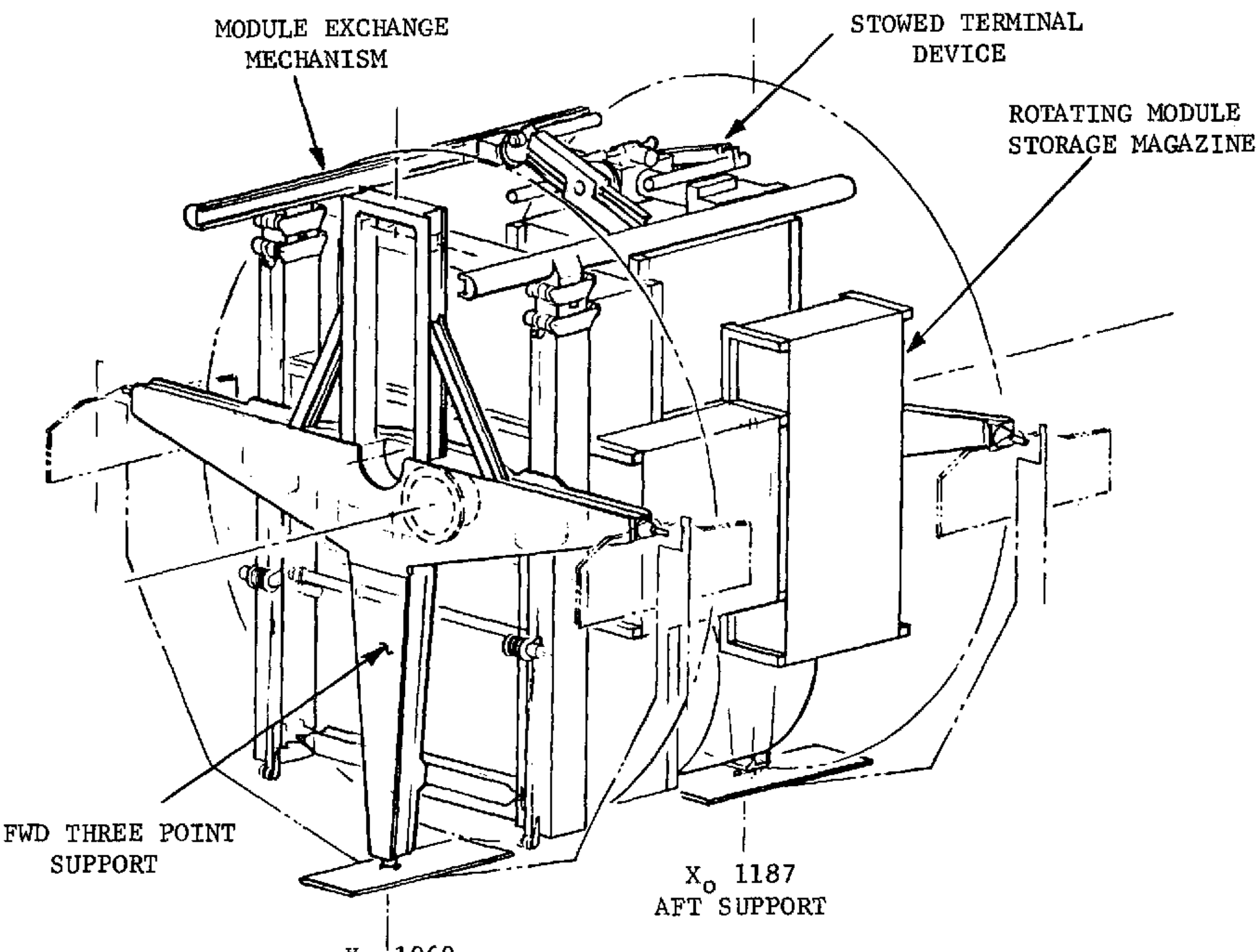

FIGURE 2-16 SHUTTLE SPMS INSTALLATION

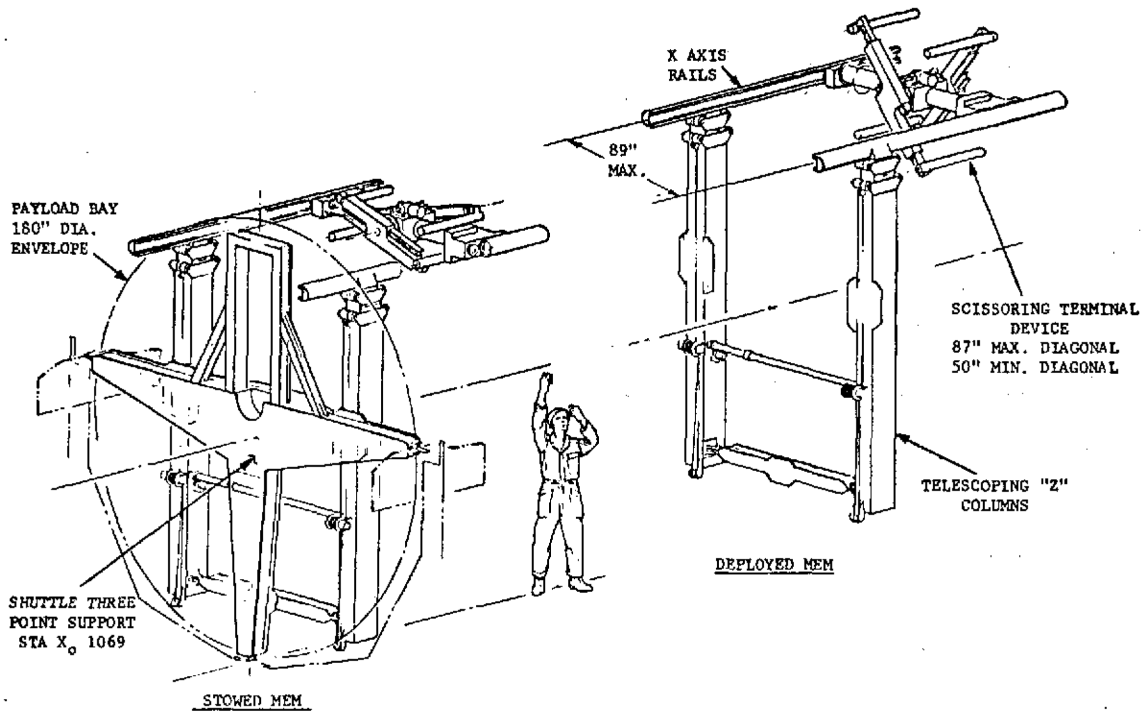

FIGURE 2-17 SPMS MODULE EXCHANGE MECHANISM (MEM)

During the module exchange operation the double-ended terminal device first releases and holds the replacement module from the magazine on one set of end effectors. The device is repositioned and next extracts the used module from the spacecraft. The assembly is then rotated 180 degrees to install the replacement module. The used module is then returned to the module magazine occupying the spot vacated by its replacement. This concept eliminates the need for a separate module holding fixture for exchange resulting in simplified module replacement procedures.

The modules are stored in the magazine during launch and retrieval and the magazine is insulated on all surfaces except the outboard module faces. Module remote connectors are mated to magazine mounted connectors to permit module monitoring and to provide heater power to the modules during storage. The modules are attached to the magazine by the corner latches and are removed and replaced by the Terminal Device as described above.

SPMS characteristics taken from the SPAR report are summarized in Table 2-3.

Table 2-3 SPMS Characteristics

Working Stroke	130 in. in X-Axis 214 in. in Z-Axis 40 in. in Y-Axis
Tip Force	300 lbs. through 18 in. Travel
Stiffness of Structure	230 lbs./in. (At Full Extension)
Precision (No Load)	$\pm$ 0.25 in.
Speed of Operation	1 in./sec. (Unloaded) 0.10 in./sec. Module Engage Under 300 lb. load
Stopping Distance	0.25 in. at 1 in./sec. with 900 lb. Mass
Dexterity & Control	4 DOF, Force Feedback Control, Visual Position Sensing
Storage Capacity	Up to 9 Spacecraft & Instrument Modules
Weight	2840 Lbs.
Operational Power	250 Watts
Cycle Time	15 Minutes Nominal
Flight Environments	Shuttle Launch and Orbit

Replaceable items not handled by the SPMS are the Solar Array and TDRSS Antenna assembly, which are removed and replaced by the SAMS manipulator, and are stowed in the Cargo Bay forward of the Retention Cradle. The stowed array and antenna are supported by a Storage Fixture at Shuttle Station 715 and the Retention Cradle at Station 951.

2.5.3 SPACECRAFT RESUPPLY PROVISIONS

EOS spacecraft provisions for resupply are shown for the Resupply Configuration on Figure 2-18. Note that this configuration also incorporates the retrieval features previously described thus providing either resupply or retrieval capability.

Major resupply provisions are:

(1) Replaceable ACS, Power, and C&DH subsystem modules using SPMS.

(2) Replaceable TM and HRPI (or other designated payload) modules by SPMS.

(3) Exchange Wideband Module including gimballed antennas with SPMS.

(4) Exchange Solar Array and TDRSS with SAMS. Both appendages refold for storage.

(5) Axial exchange of Propulsion Module using modified Docking Frame and SPMS.

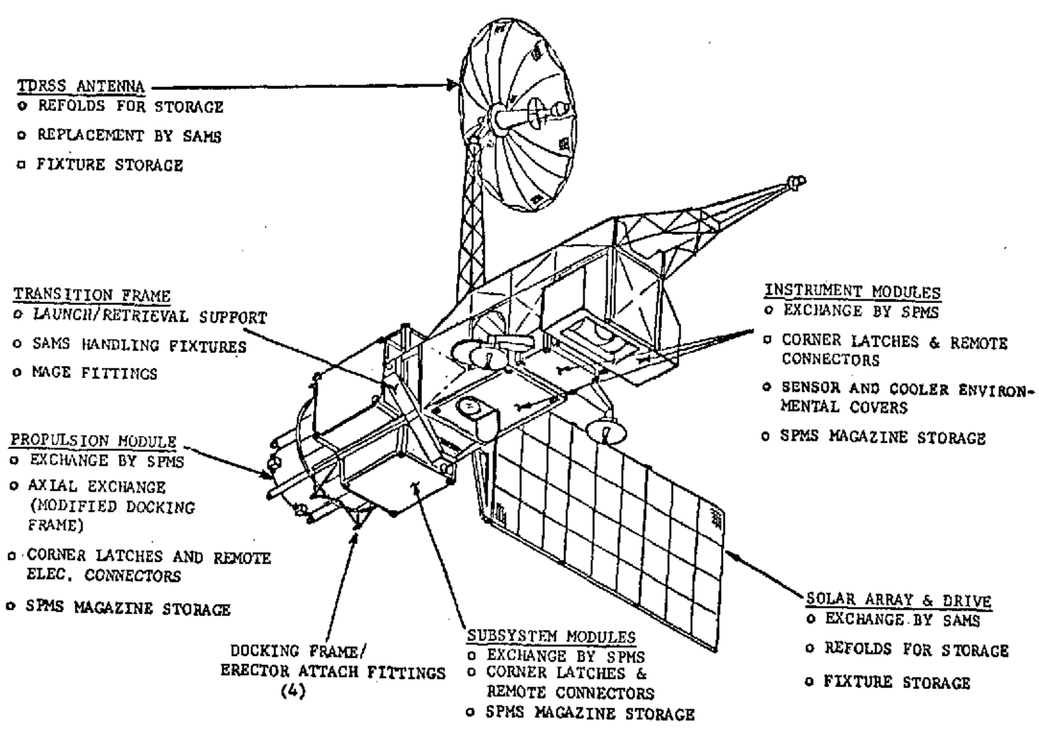

FIGURE 2-18 EOS SPACECRAFT RESUPPLY PROVISIONS

Module corner latches use a common design varying only in length. All subsystem modules, the Wideband Module and the Propulsion Module use the 18-inch long latch shown on Figure 2-19, and the deeper instrument modules use longer latches as required. The latch design shown is a variation on the original GSFC latch design with modifications designed to reduce cost and weight of the unit.

An ACME threaded stud and a conical seat are located on the spacecraft structure at the four module corners. The latch attached to the module consists of an elongated nut and a male spline assembly, which is soft spring loaded in its normal axial position. The spline engages the upper shaft which is supported by bearings and has at its outer end a knob which interfaces with the exchange mechanism terminal devices. When a module is being installed in a shuttle operation, the conical insert at the base of the latch provides a guiding action over the pointed contour of the stud.

As the module is forced against the structure the ACME nuts recede into the latch against the soft spring pressure until such time as the outer shaft is rotated by the MEMS terminals. When the nuts are completely torqued, the required mounting force of 3000 lbs is present. This approach accommodates the condition where only two terminals engage

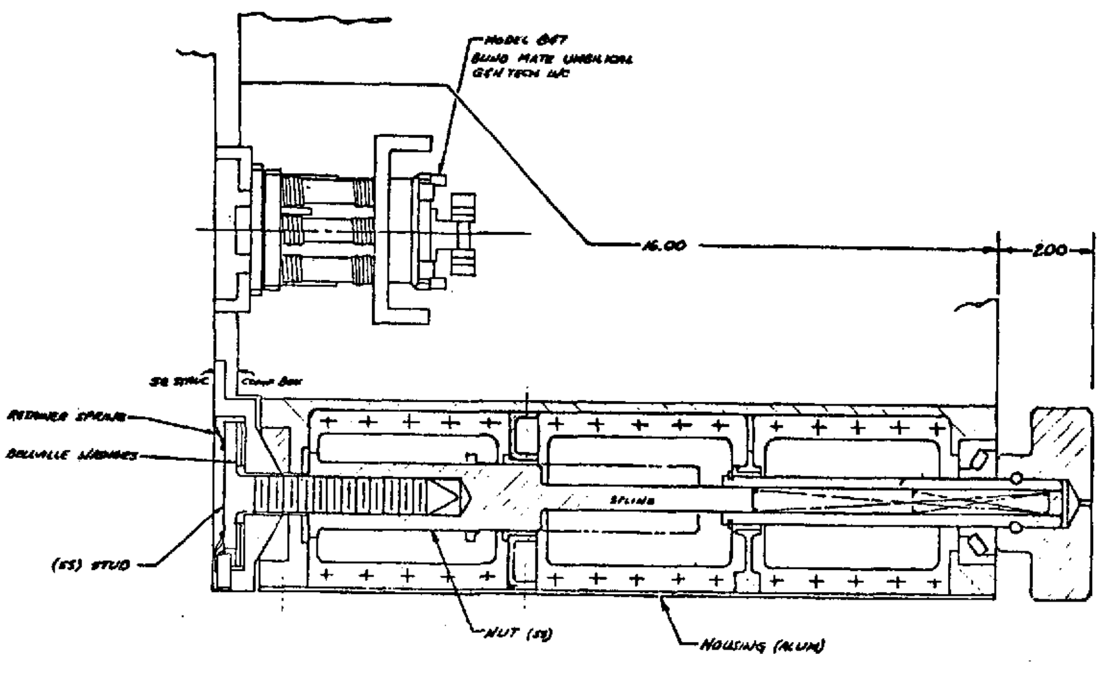

FIGURE 2-19 EOS MODULE LATCH MECHANISM

the module at one time. If four terminals are available, it may be possible to implement further design simplification. Note that the corner guide rails have been eliminated to save weight and the mounting stud and conical seats have been configured to accommodate the $\pm$ .25-inch MEM positioning accuracy.

Weight for the 18-inch latch including the fixed stud has been estimated at eight pounds per corner or 32 pounds per module.

The connectors which form the electrical interface between the module and the spacecraft are required to automatically mate as the module is installed.

The mechanism, shown in Figure 2-20, is a blind mate umbilical, manufactured by G&H Technology, Inc., currently being qualified for the F-14 weapon rail, and appears to be a good candidate. The device will allow a $\pm$ 0.15" misalignment at mating. Mating and disconnect forces are from 100 to 185 lbs. The device would be located near a corner latch as shown so that the forces generated by the ACME screw in both mating and demating would be directly transmitted to the connector without undue moments on the module. The connectors mate after the mounting studs are engaged positioning the

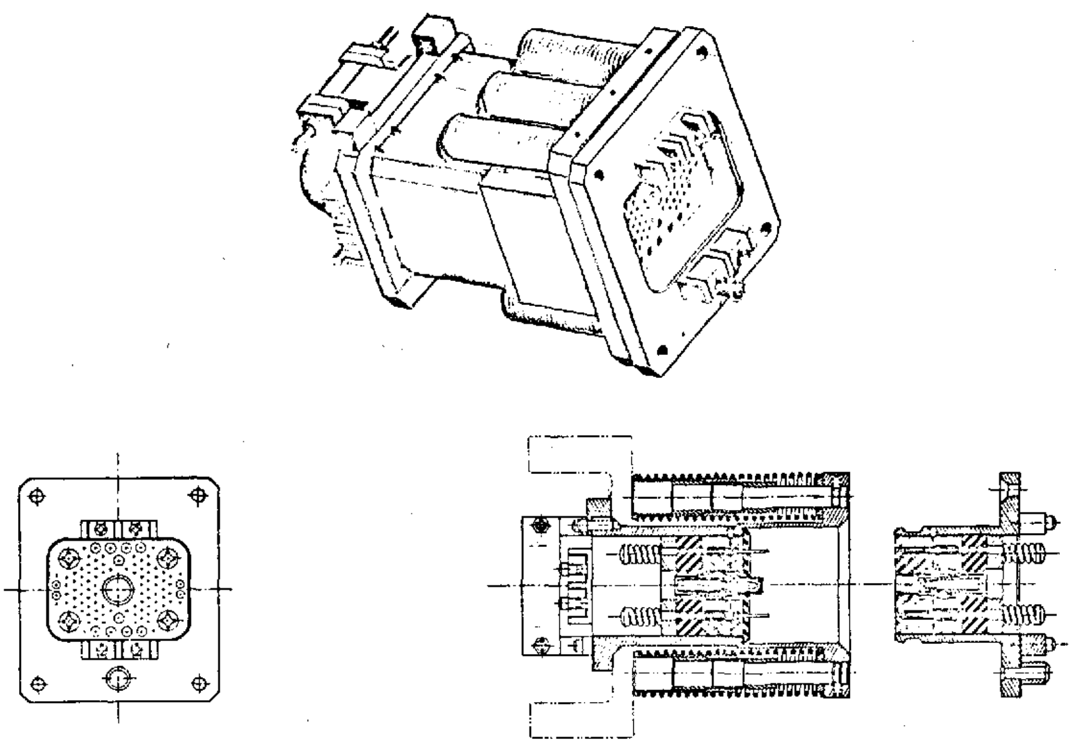

FIGURE 2-20 MODULE REMOTE ELECTRICAL DISCONNECT

connector halves well within the $\pm .15$ inch misalignment allowance.

Instrument modules are designed to house and support instruments and vary in size and mount configuration to meet the unique requirements imposed by each instrument. The module shown on Figure 2-21 for the Thematic Mapper and HRPI are typical instrument module designs. The basic structure is a welded 6061 aluminum frame configured to transfer the three-point instrument mount loads to the four inboard corner reaction points. The latch mechanisms are located at each module corner and electrical disconnects on the inboard module surface near a corner.

The modules are completely insulated except for the earth viewing heat rejection surfaces and internal guard heaters are provided to maintain temperatures during module storage or orbital operations. Note that one side of the TM module is open to accommodate the TM cooler cover door.

Subsystem modules for resupply are identical to the non-resupply design except for incorporation of the four corner latches and remote electrical disconnects. Construction and arrangement of a typical module is shown on Figure 2-22.

The Propulsion Module for resupply, Figure 2-23, fits within the central cavity formed by the subsystem support truss structure and is attached to a fixed cylindrical skirt by four module latches as shown. This module is designed for axial removal for replacement which is accomplished by a modified Docking Platform and the SPMS Module Exchange Mechanism as shown on Figure 2-24. The Platform changes consist of providing a center pivot mechanism for the Docking Latch support arms which will rotate the spacecraft to a horizontal position after erection by the FSS erection mechanism. This rotation positions the Propulsion Module in the proper orientation for extraction by the MEM Terminal Device. This proposed modification to the FSS has been presented to RI for evaluation and appears to be a completely feasible concept.

A layout of the SPMS Module Magazine housing the full EOS module complement and stowed Terminal Device is shown on Figure 2-25. The basic magazine design and interfaces remain unchanged and as shown, the magazine with all modules installed is within the 180-inch diameter cargo bay envelope, and the 97-inch length allocated for the magazine by RI.

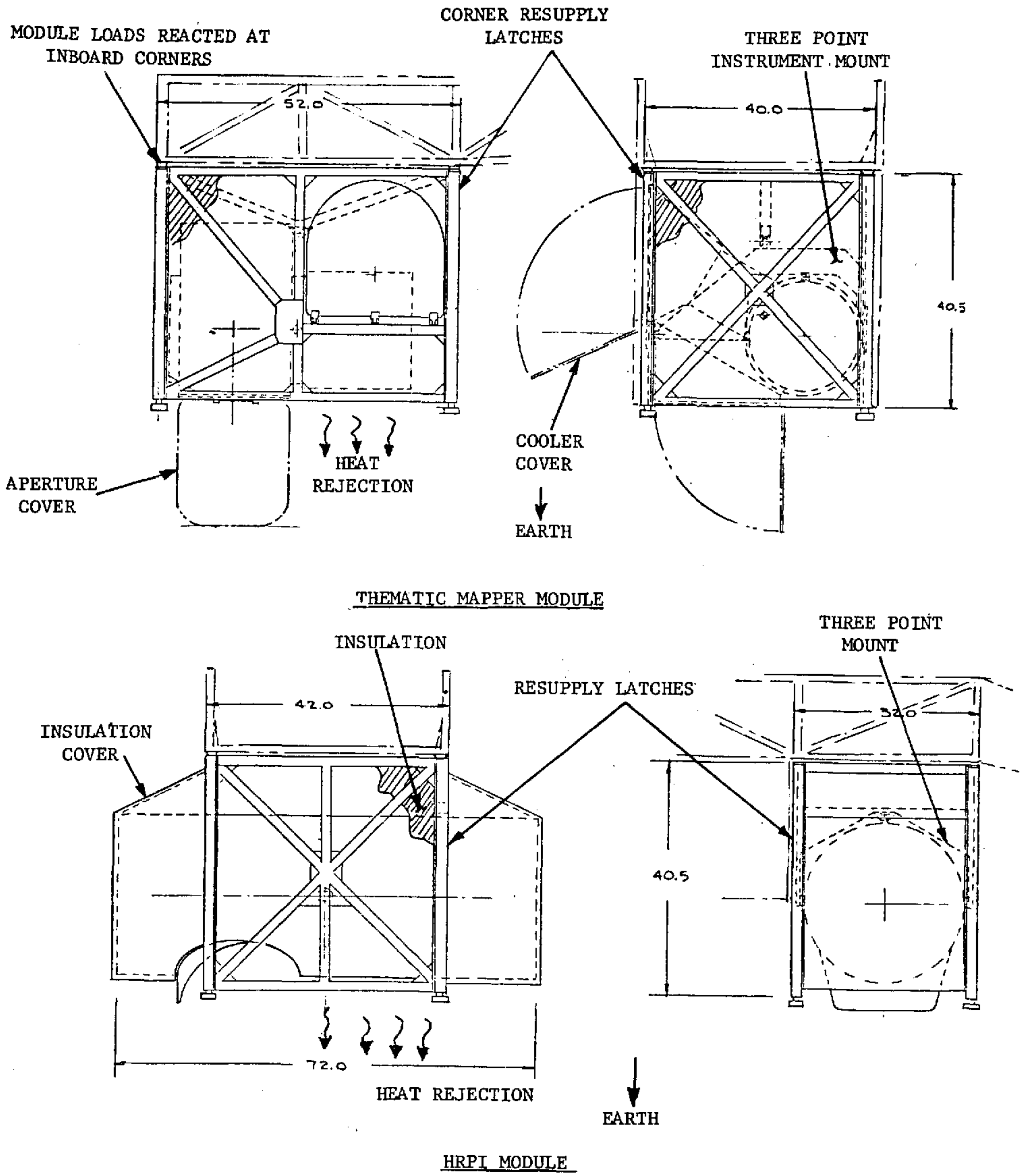

FIGURE 2-21 EOS INSTRUMENT MODULE ARRANGEMENTS

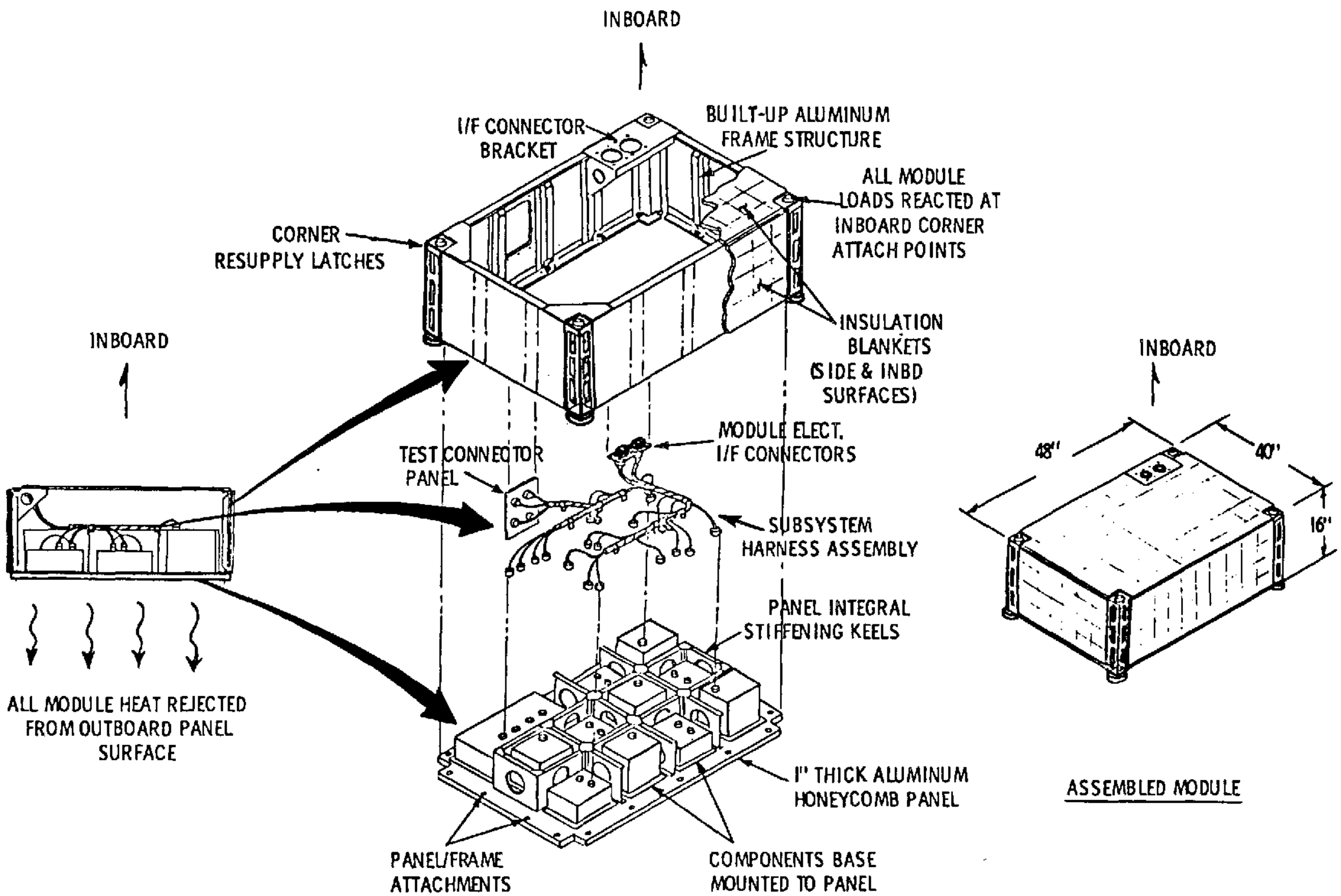

FIGURE 2-22 EOS RESUPPLY SUBSYSTEM MODULE

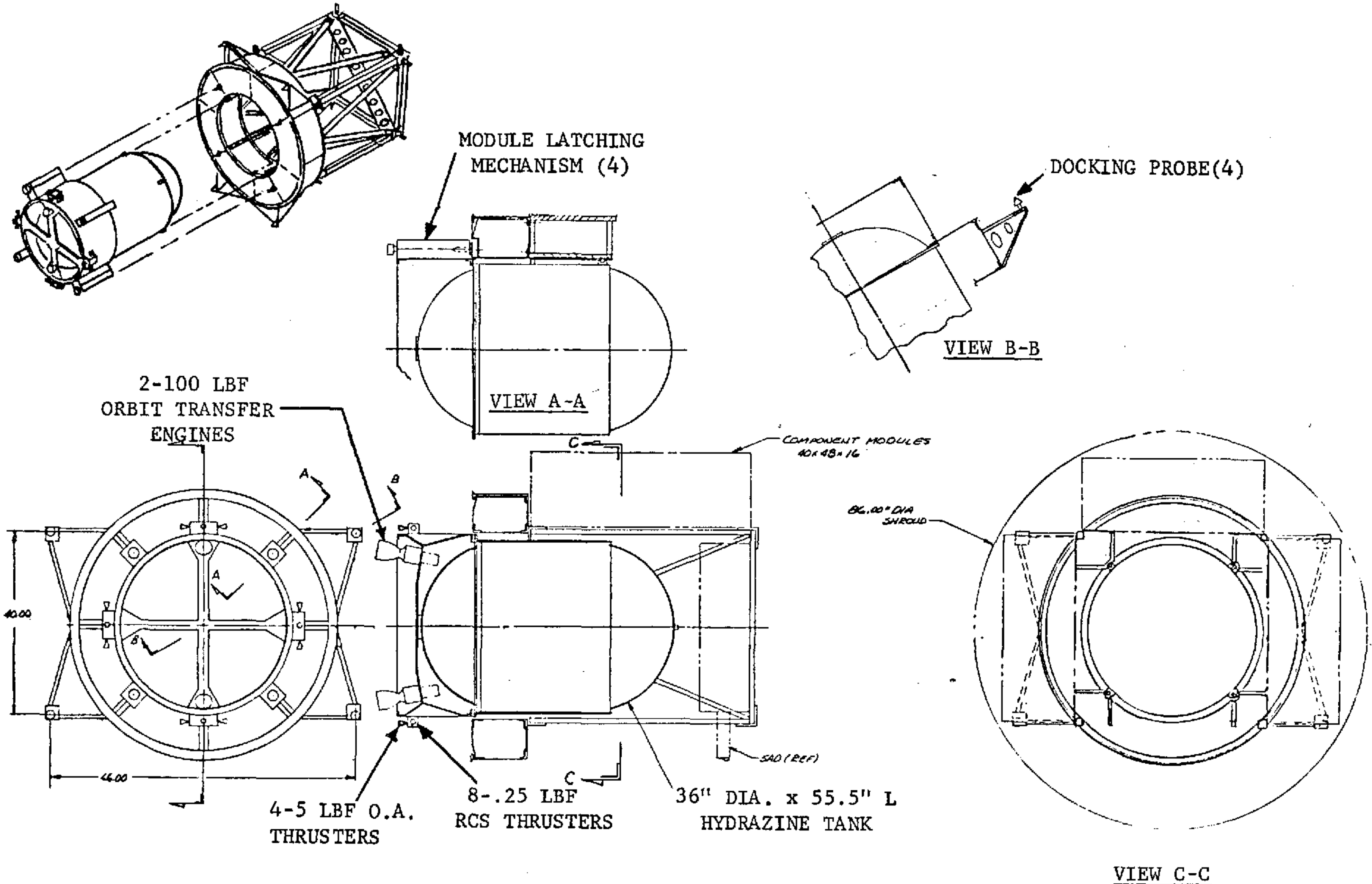

FIGURE 2-23 EOS RESUPPLY PROPULSION MODULE

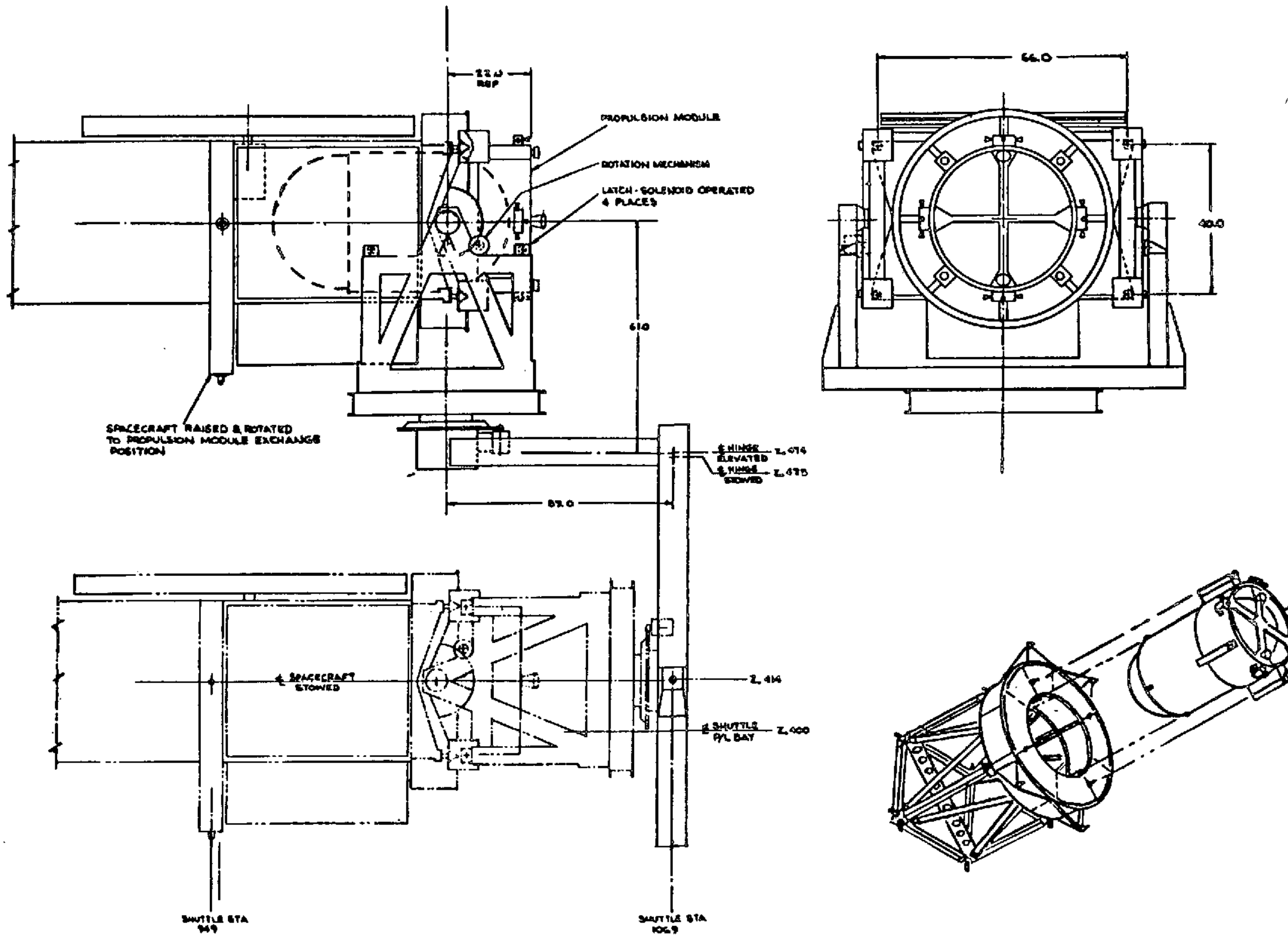

FIGURE 2-24 EOS AXIAL PROPULSION MODULE RESUPPLY

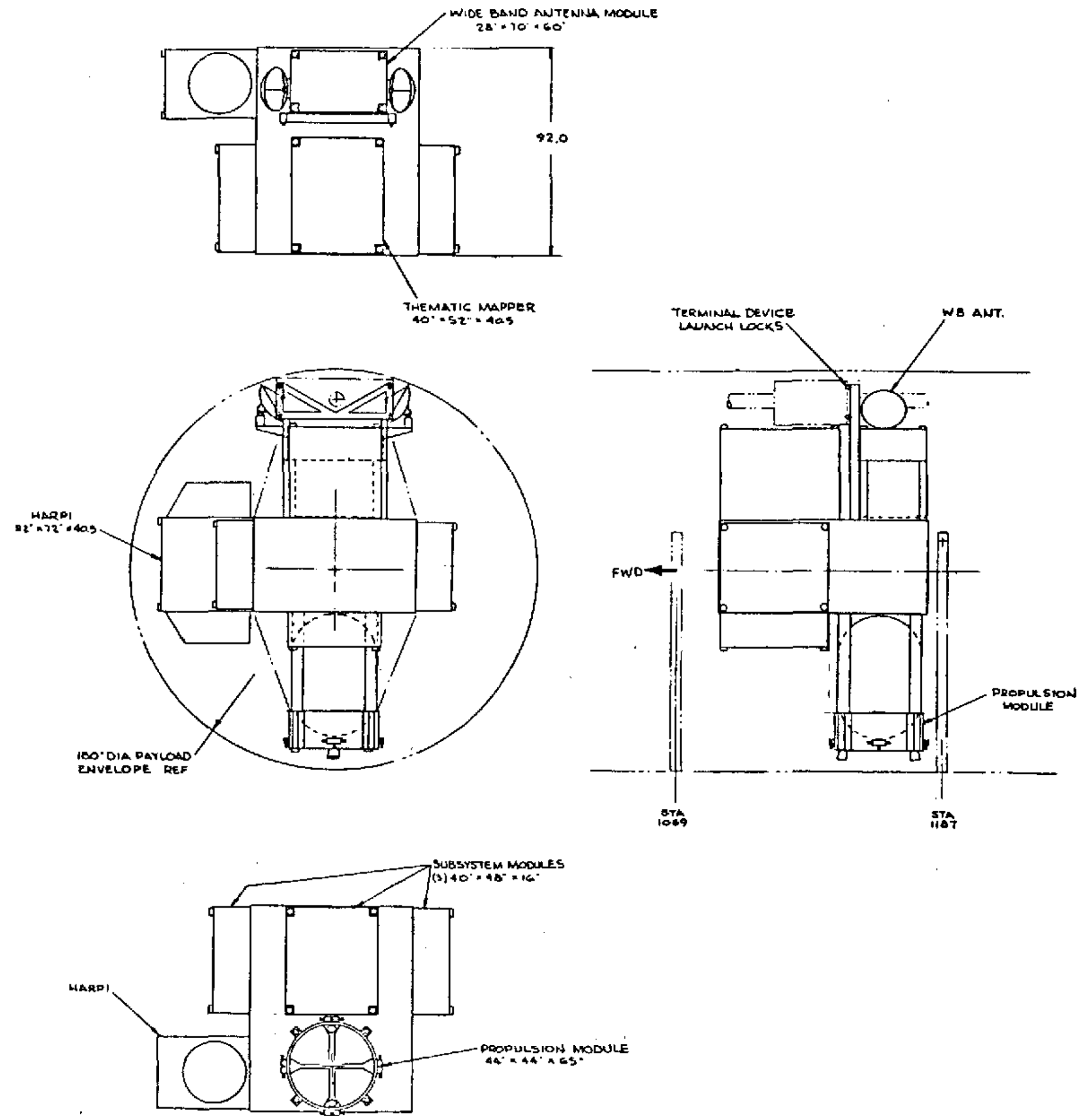

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FIGURE 2-25 SPMS MAGAZINE LAYOUT

2.5.4 ALTERNATE SECTION RESUPPLY

An alternate resupply mode is illustrated on Figure 2-26 using the TM and Dual MSS Configuration Spacecraft. The spacecraft is shown installed in the FSS Retention Cradle and held at the Transition Frame. The Transition Frame has forward and aft latches at the Section attach corners which are activated by SAMS interface torque heads on the Transition Frame driving internal linkages to release either set of four latches independently.

Guide rails and axial translation mechanisms integral to the cradle would separate either the BUS or Instrument Sections from the Frame. The Section would then be lifted from the rails by SAMS and placed on a temporary holding fixture, after which SAM would place a replacement Section on the rails for subsequent mating to the Frame attachments. The used Section would then be stowed in the replacement modules place for retrieval.

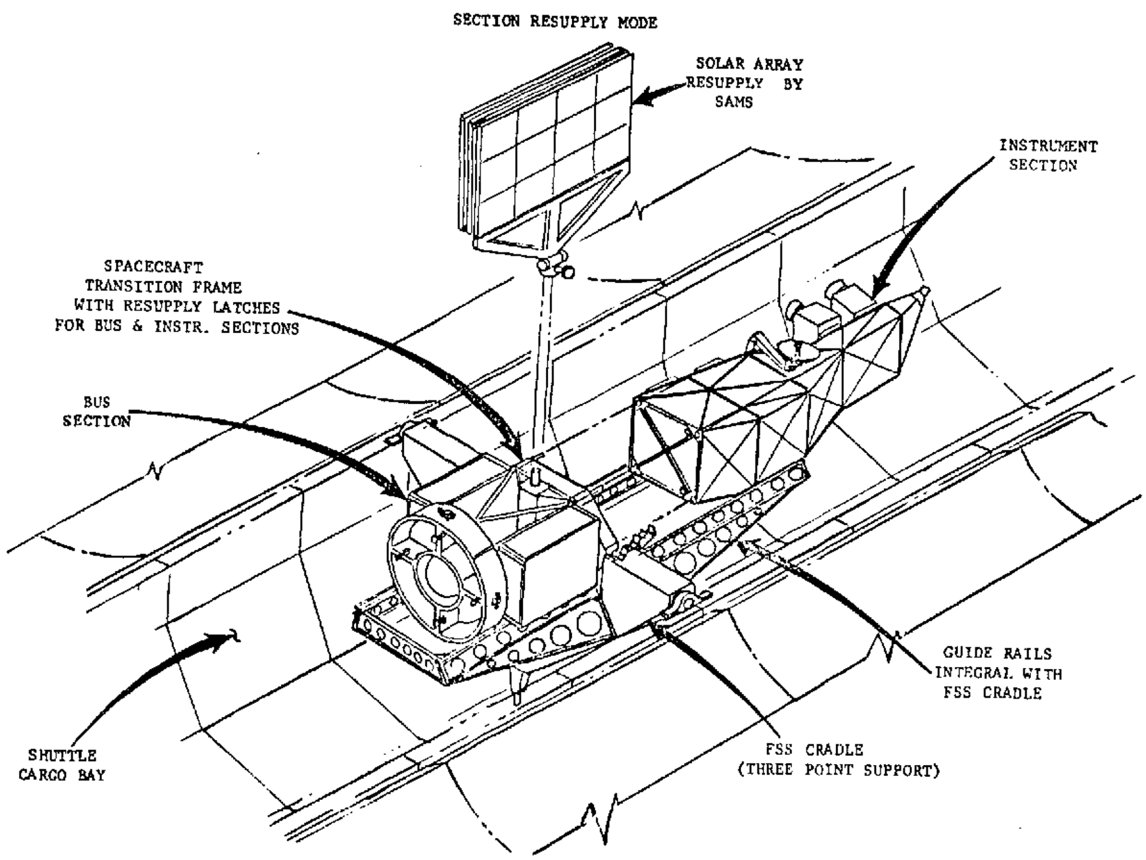

FIGURE 2-26 EOS SECTION RESUPPLY CONCEPT

The FSS could incorporate an erection/docking mechanism if required, or SAMS could possibly be employed for both deployment and capture and retrieval stowage. A layout of the Shuttle cargo bay for this mode showing locations of the Frame, spacecraft, and replacement Sections is presented on Figure 2-27.

This resupply mode results in a simplified exchange mechanism, requires less weight and volume to the spacecraft and could be used with Delta launched spacecraft configurations. It is recommended that this method of resupply should be investigated in more detail since it provides the capability to make significant changes in the instrument payload without requiring spacecraft retrieval.

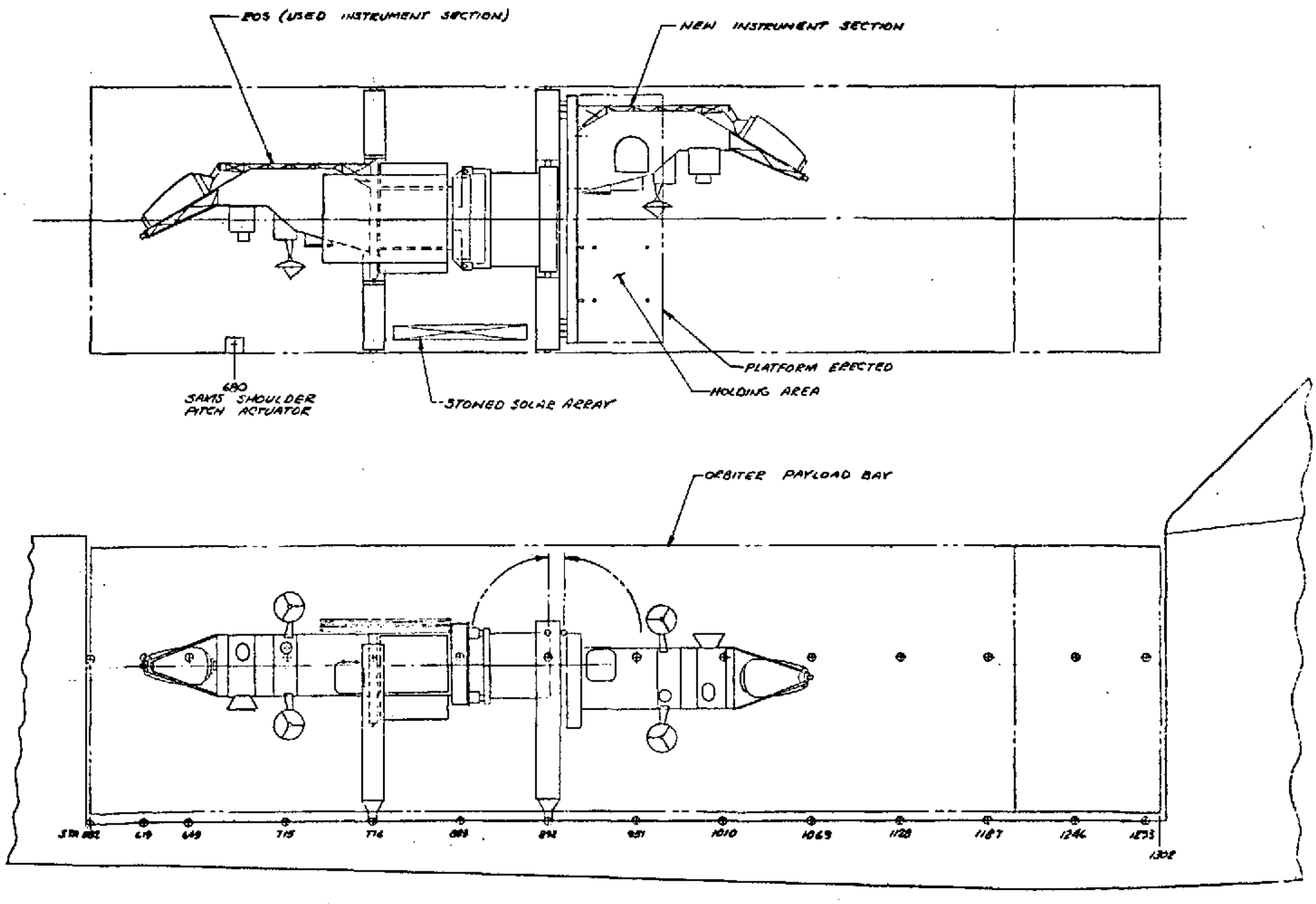

FIGURE 2-27 EOS SECTION RESUPPLY CONCEPT - CARGO BAY LAYOUT

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2.5.5 RETRIEVAL/RESUPPLY SUMMARY

A comparison of spacecraft retrieval, module resupply and section resupply impacts on the spacecraft and Shuttle systems is shown in Table 2-4, and a breakdown of spacecraft weights for retrieval and resupply provisions is shown in Table 2-5.

TABLE 2-4

EOS RETRIEVAL/RESUPPLY SUMMARY

MODE	SPACECRAFT DESIGN	SHUTTLE EQUIPMENT	CONCLUSIONS
Retrieve **(no resupply**	• Fixed S/S modules • Built in instrument mts. • Transition frame • S/S & instrument sections rigidly joined at frame • 150 lb Δ weight	• Simplified FSS (retention & erection only) • SAMS for S/C capture	• Retrieve for either S/S or instrument failure • Ground repair • Simplified Shuttle interfaces & equipment • Lowest weight S/C for re-use
S/S (BUS) and Instrument Sections Resupply (& S/C retrieval)	• Fixed S/S modules • Built in instrument mts. • Transition frame • S/S & instrument sections removable at transition frame • 255 lb Δ weight	• FSS modified to incorporate section exchange mechanisms (horizontal exchange) • SAMS for handling sections • Storage fixtures	• BUS or instrument section exchange on Shuttle • Maximum shuttle payload sharing • Good for major S/S or instrument changes • Moderate weight and cost
Module Resupply (& S/C Retrieval) **BASELINE**	• Removable S/S and instrument modules with remote latches & elect. disconnects • Transition frame • S/S & instrument sections rigidly joined at frame • Replaceable appendages • 600 lb Δ weight	• FSS including S/C indexing capability • SPMS for module exchange • SAMS for appendage exchange • Storage provisions for modules in SPMS magazine & fixtures for appendages	• Exchange failed S/S module or instrument • Most complex & heaviest spacecraft • Requires most complex exchange mechanisms • Maximum utilization of Shuttle • Potentially most cost effective

The summary indicates that the module resupply mode results in the highest weight impact to the spacecraft and also requires added volume over the Retrievable or Section Resupply designs, which would require a Shuttle or Titan L/V initial launch at higher cost. The resupply mode, however, can potentially produce the maximum cost effectiveness for the overall system during the Shuttle era. The Retrieve and Section Resupply modes result in significantly lower spacecraft weight and volume and are desirable for early EOS missions launched by the low cost Delta booster. The modular spacecraft design is completely compatible with any of these resupply modes possessing the design flexibility to configure for retrieve only for early applications and incorporate resupply on later missions.

TABLE 2-5

WEIGHT BREAKDOWN
FOR
SPACECRAFT RETRIEVAL/RESUPPLY
PROVISIONS

RESUPPLY MODE	SPACECRAFT Δ WEIGHT
I. RETRIEVE ONLY	
• Handling Provisions	30#
• Transition Frame	120#
TOTAL RETRIEVE	150#
II. SECTION RESUPPLY (+ RETRIEVAL)	
• Retrieve Total	150#
• Resupply Latches (8)	80#
• Electrical Connectors (8)	25#
TOTAL SECTION RESUPPLY (INCL. RETRIEVE)	255#
III. MODULE RESUPPLY (+ RETRIEVAL)	
• Retrieval Total	150#
• Added Handling Provisions	50#
• Resupply Latches (34)	250#
• Elect. Disconnects (16)	50#
• Module Structures (2)	100#
TOTAL RESUPPLY (INCL. RETRIEVE)	600#

SECTION 3

ELECTRICAL INTERFACES

3.1 INTRODUCTION

An electrical interface is maintained between the EOS spacecraft and the shuttle orbiter during three basic modes of operation:

1. Attached in the retention cradle during pre-launch, ascent or return phases of the mission
2. Attached to the positioning platform immediately prior to and during servicing
3. Detached

The attached modes provide hardwire connections for all spacecraft input and output signals: power, command, telemetry and data. The detached mode employs RF communications between the spacecraft and orbiter, with the spacecraft receiving power from its own solar arrays and batteries. This mode is used to provide a check of the entire spacecraft while the orbiter is on station and is used in lieu of direct communication with the ground or via TDRSS. This section discusses the design considerations for the electrical interfaces to achieve spacecraft-shuttle compatibility.

3.2 POWER INTERFACE

Since the EOS spacecraft may be stowed in the orbiter cargo bay for as long as several days, the spacecraft will require shuttle power to support checkout and heating needs. Further, the batteries should be fully recharged at the time of spacecraft separation from the orbiter to cope with the power demands associated with the spacecraft transfer to a higher orbit and any subsequent solar and attitude acquisition maneuvers. Hence power must be supplied by the orbiter during the stowage phase.

In addition to power provided by ground support equipment, the orbiter can provide up to 50 kilowatt-hours of electrical energy that is not chargeable to the payload. This power, limited to 1 kwatt average and 1.5 kwatt peak during ascent (descent) and 7 kwatt average and 12 kwatt peak on orbit, is sufficient for the spacecraft needs. Heating requirements will have to be more fully examined if the cargo bay doors are to remain open in orbit for long periods of time.

The power from the orbiter is supplied at 27 to 32 VDC and is available at a disconnect located at station 695. An additional harness segment contained as part of the Flight Support System (FSS) routes the power to a convenient automatic mating connector where the spacecraft is secured to the FSS. This connector mates to the spacecraft umbilical connector, which provides the necessary power, grounding, and control interfaces.

Several methods have been considered for conditioning the unregulated orbiter power (27 to 32 VDC) to be compatible with the regulated power of the EOS spacecraft (28 VDC $\pm$1%). In one method, shown in Figure 3-1, a buck-boost regulator is used to match the orbiter raw power with the regulated output of the spacecraft power system. To avoid any overlap of control sensitivity with the spacecraft regulation controls, the buck-boost system is designed to respond to the same driver signals developed in the power system central control unit. In normal spacecraft operation the driver signals, which are proportional to bus voltage error, are used to sequentially modulate the solar array shunt regulator, the charge regulators and the discharge regulators. The

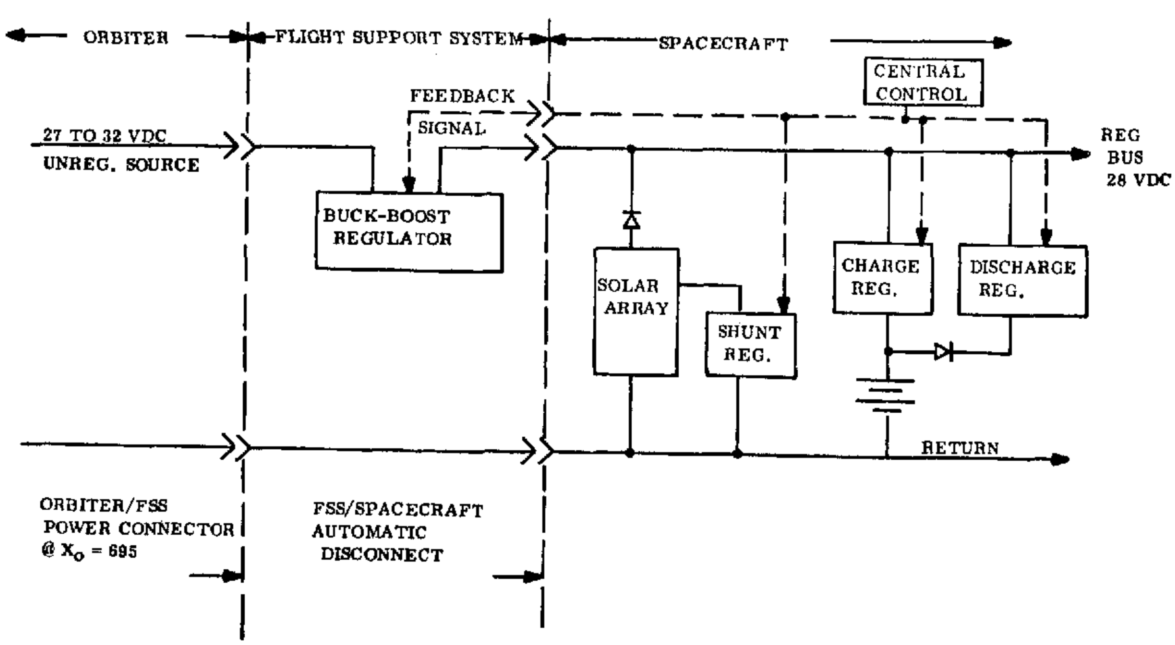


FIGURE 3-1 POWER INTERFACE CONCEPT - METHOD NO. 1

modulation relationship is illustrated on Figure 3-2(a). Positive error signals, indicating excessive voltage, are used to modulate the shunt regulator to a full-off position as the bus voltage decreases. As the voltage drops further battery charging is inhibited to divert available power to the loads. As it drops still further the boost regulator is turned from full-off to full-on drawing upon battery power to fill the load demands. In applying the buck-boost regulator to this system the same driver signals used for the shunt regulator would be used for the buck-boost system as shown on Figure 3-2(b). Where a high voltage error signal fully turns on the shunt regulator, it would fully turn-off the buck-boost regulator. Both actions are consistent in limiting source power in response to a high bus voltage although modulation of the shunt regulator has little effect on system output since the solar array is largely inactive during the stowage phase. As the voltage drops the buck-boost regulator would be turned on more and more to a full-on condition. A current limiting device would be used to limit the peak power drawn from the buck-boost regulator to about 20 amperes. Further demands, such as may occur during checkout of large loads, would be supplied

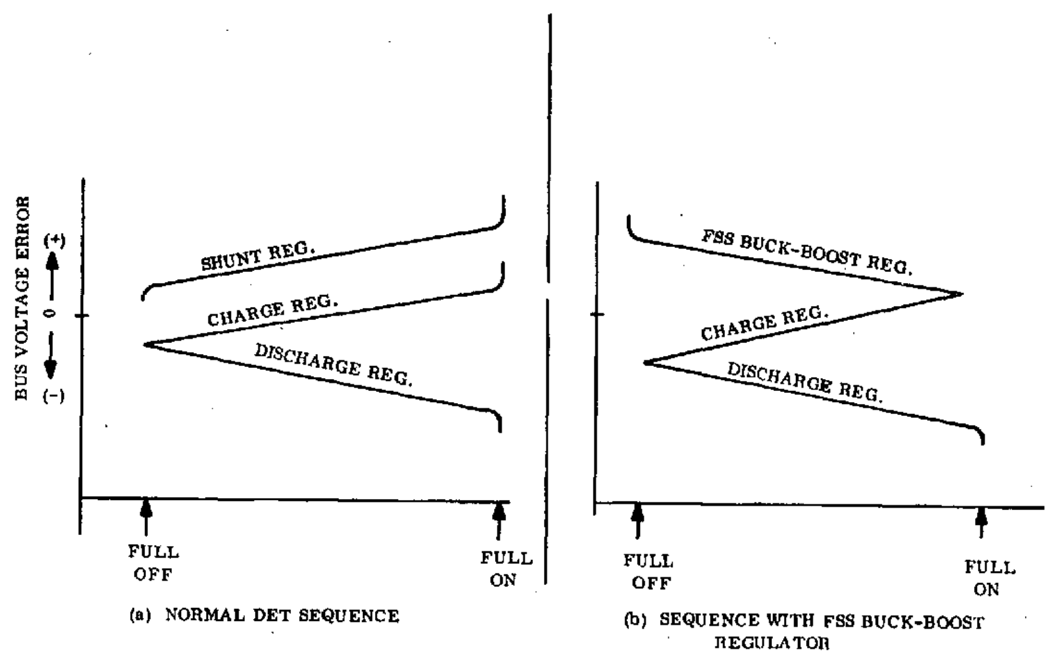


FIGURE 3-2 POWER SYSTEM LOGIC SEQUENCE

from the batteries through their boost discharge regulators in response to a low voltage driver signal. The method described above requires the insertion of a conditioning function (buck-boost regulator) between the orbiter power source and the spacecraft power system. This would be located with the FSS and remain with the orbiter.

A second method eliminates the buck-boost regulator but requires the addition of several switches to the spacecraft power system. This insertion of switches is shown on Figure 3-3. The direct flow of source power to the loads is interrupted; all source power is routed through the charge regulators and is used either to charge the batteries or flow directly to the input side of the boost discharge regulators. Because of the interruption, the source power may be unregulated over a range of about 26 to 40 volts determined on the lower end by the number of battery cells used and on the upper end by the circuit design of the charge regulators. The 27 to 32 VDC orbiter limits fall comfortably within this range.

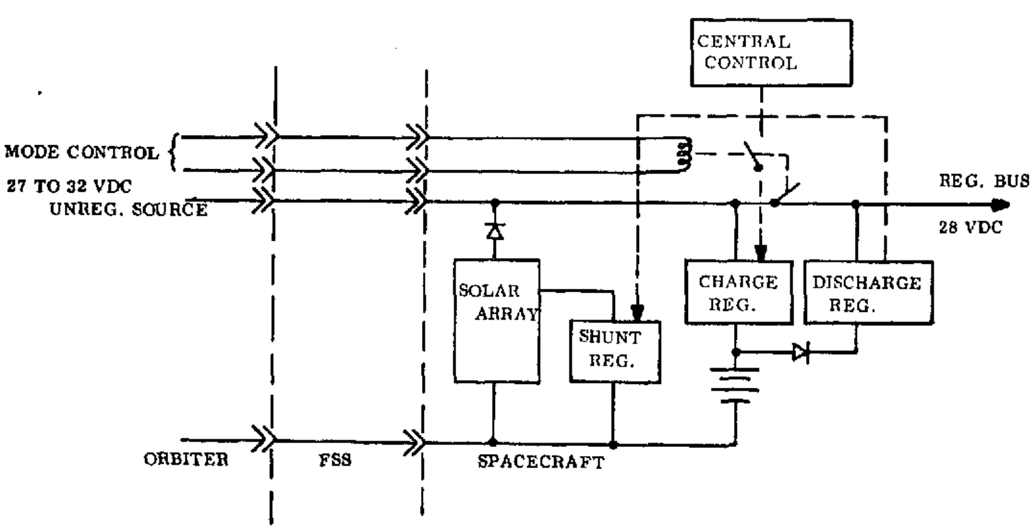


FIGURE 3-3 POWER INTERFACE CONCEPT - METHOD NO. 2

Because the logic of the normal DET system requires that the charge and discharge regulators cannot operate simultaneously, it is necessary to interrupt the inhibit signal to the batteries from the central control unit. Without the inhibit signals, the charge regulators operate only in response to battery charge parameters which is the desired mode for the arrangement as described.

Grounding practice usually dictates that power and signal returns be grounded at one point of the vehicle structure. This approach is specified for the shuttle orbiter and will likewise be specified for the spacecraft. When the spacecraft is attached to the orbiter, the concept of a unipoint ground is violated since in actuality two separate grounding points exist -- one in the orbiter and one in the spacecraft. To eliminate the possibility of voltage drops resulting from current flows through the mated structures from one ground point to the other, a low resistance path between the ground points should be established. Sufficient connector pins in the interface connectors of the orbiter and spacecraft should be allocated for this purpose.

Capability should be provided for completely isolating the spacecraft power system and loads from the shuttle electrical system during the time that the spacecraft is mated to the orbiter. This is necessary for safety reasons in the event that shorts have developed in either the power system or loads. Isolation switches should be provided for the solar array, each battery and at the main distribution bus to the spacecraft loads. The isolation switches should be operable from the shuttle orbiter with power derived from the orbiter.

As a matter of good practice, all signal and power lines through the orbiter/FSS/ spacecraft interface should be isolatable on the spacecraft side to deadface the spacecraft umbilical disconnect. This can be accomplished effectively by using holding relays in each line which are held in contact from the orbiter side of the interface and automatically open after umbilical disconnect separation. This practice is presently used on the Nimbus and ERTS spacecraft.

Retrieval of the spacecraft by the orbiter may occur for reasons of refurbishment and repair and may involve hazards as a result of damaged or faulty equipment. Shorted batteries, equipment or loads could result in high current conditions once power from the orbiter is applied to the spacecraft after mating. The signal and power isolation switches mentioned in the previous paragraph can serve to permit mating with the umbilical disconnect without the danger of dealing with live circuits. Once mated the caution/warning system can be used to determine system health before critical circuits are activated. If necessary, key segments of the power system can be isolated earlier since they would be activated from the orbiter side as explained earlier. The important factors in this regard are to recognize the precedence of crew safety and to provide sufficient operational flexibility from the orbiter side to eliminate any hazardous procedures.

3.3 COMMAND AND TELEMETRY INTERFACE

3.3.1 COMMAND

Command data to the EOS may be initiated at two sources: (1) on-board the orbiter and (2) at the ground with the orbiter acting as a relay. During the period of time that EOS is attached to the retention cradle (pre-launch, ascent, or return) or to the positioning platform (on-orbit), these data are transmitted hardwire to the spacecraft umbilical from the signal interface panel located at Station 576 on the orbiter bulkhead. When EOS is detached from the orbiter, these commands may be transmitted by the orbiter S-band payload communication link.

Commands initiated on-board the orbiter are obtained from stored data in the orbiter's computer which is called up by the crew via a keyboard. These data are formatted to look like ground initiated commands and are used as a contingency during periods of time ground commanding is not available. As such, only a limited number of critical control commands will be available for this mode of operation. Ground initiated commands are received by the orbiter S-band communication link (direct or via TDRSS). These commands are decoded and validated prior to transmission to the payload. If the message received is in error, the error message is transmitted to the ground station. The payload has the option of having the erroneous message rejected or passed on to it. EOS will only accept good data to avoid possible erroneous activation of critical functions while attached to the shuttle.

Command data to the payload are formatted as shown in Figure 3-4. The first four bits identify the orbiter address. The next four bits identify which of several possible payloads is being commanded. The next eight bits provide a unique sync pattern for

4 Bits	4 Bits	8 Bits	32 Bits	1 Bit	1 Bit	77 Bits	1 Bit
Orbiter Address	EOS Address	Sync	CMD Data	Fixed Data Zero	Fixed Data Zero	BCH Code	Fixed Data Zero

FIGURE 3-4 SHUTTLE COMMAND FORMAT

use by the payload in decoding the following 32 bits of command data. Bits 49 and 50 are zero set, followed by 77 bits of BCH (Bose-Chaudhuri-Hocquenghem) code data to form a 127,50 block code on the initial 50 bits. The 128th bit is zero set. Data are transmitted to the payload as biphase-L at an 8 kbps rate. This permits an information rate of 2 kbps, or 50 commands per second, the same as the EOS uplink command rate. However, additional decoding is necessary on-board EOS to convert the shuttle format to that compatible with the EOS central command decoder (CCD).

EOS will provide this decoding using a special software applications package in the OBC. This package will utilize a dedicated direct memory access channel tied to the EOS umbilical via the OBC I/O interface. (NOTE: For RF commanding from shuttle in the detached mode, this interface must previously be switched to the central command decoder output of the modulation processor.) The OBC will accept the incoming 8 kbps data and extract from each 128 bit word the 32 bits which represent encoded EOS command data. These data are either reformatted as 32 bit words for application to the supervisory data bus via the telemetry format generator or (if memory load or delayed command) are transferred to an appropriate section of the OBC memory for later processing.

3.3.2 TELEMETRY

The orbiter is capable of receiving narrowband telemetry data from the EOS via a hardwire interface between the EOS umbilical and the orbiter signal interface panel located at Station 576 or an S-band RF link. The hardwire link will be used during the period of time EOS is attached to the retention cradle or positioning platform and is the primary mode of operation. The RF link will be used as a backup to the normal EOS communication links with STDN and TDRSS on orbit.

When EOS is attached to the orbiter, data may be input to the orbiter in either of two ways. The first permits handling of up to 64 kbps of payload (spacecraft) data and requires the payload to provide data, clock, major frame sync, and minor frame sync. Signal input timing requirements for this interface are given in Figure 3-5. The second approach is limited to 16 kbps and is dedicated to an individual payload on any one

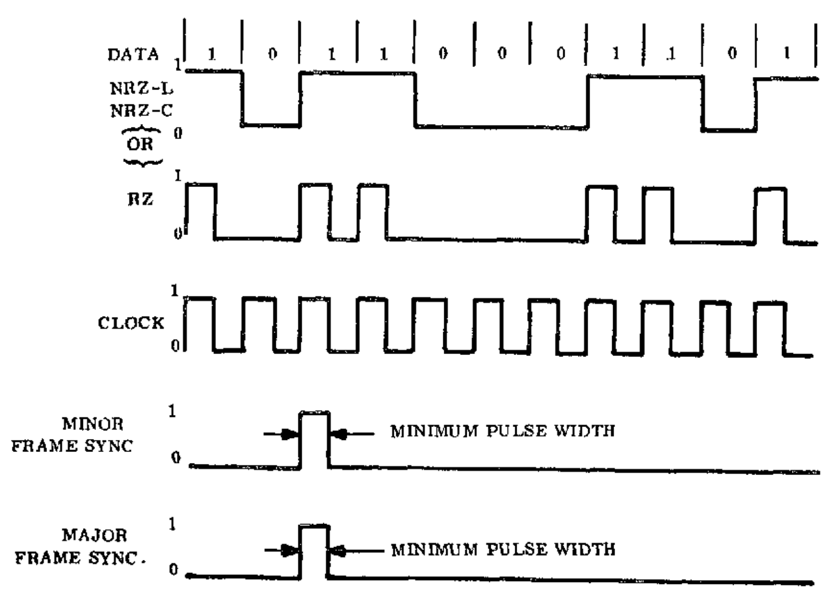


FIGURE 3-5 SIGNAL INPUT TIMING

mission. In this case, the orbiter will generate clock, major frame sync, and minor frame sync from a payload Manchester encoded input signal. The second approach for obtaining narrowband telemetry data is preferable. These data can be obtained directly from the output of the telemetry format generator via a dedicated interface in the spacecraft umbilical. The 16 kbps input limitation is adequate for the narrowband data rates currently defined for EOS and is also compatible with the RF link requirements, which are limited to 16 kbps. The above interface is acceptable only for data to be relayed to the ground or directly recorded by the orbiter. It is not acceptable for data which must be decommutated on-board the orbiter unless the orbiter is capable of decommutating the EOS format (not defined by shuttle documentation). To protect against the possibility of incompatible formats, a second interface will be provided by EOS. This interface will be tied to a dedicated direct memory access channel of the OBC. The OBC receives all narrowband telemetry data formatted by the telemetry format generator. The shuttle applications package will provide the software necessary to extract any (or all) data of interest ot the orbiter, format the data, and provide necessary synchronization for orbiter use. These data will be provided to the umbilical hardwire interface or to the modulation processor for RF transmission. It provides for flexibility adaptable to unique mission requirements. Narrowband telemetry data obtained by the orbiter may be used for on-board performance monitoring, digital recording, or direct re-transmission to the ground. Performance monitoring is used as a backup to caution and warning signals (see Paragraph 3.5) for payload status monitoring. As such, the orbiter will assess the data for out of limits conditions and/or status check against a pre-determined reference list of functions. This may be used as a convenient tool for monitoring of critical spacecraft functions during launch or descent or as a mini-functional check prior to S/C deployment on orbit. Digital recording of telemetry data, which may consist of anomaly data, periodic six second snapshot data, or be continuous may be held for subsequent analysis after the orbiter returns or may be transmitted via a shuttle S-Band FM communication link. Narrowband telemetry data may also be directly transmitted to the ground via the orbiter. In this case, it is interleaved with orbiter data and transmitted via the orbiter S-Band PM link (direct or TDRSS).

3.4 DATA INTERFACE

EOS instrument data retrieval will be limited due to shuttle capability. This capability provides a mediumband capability of 3.0 MHz analog or 1024 kbps digital data and a wideband capability of 4.0 MHz analog or 50 Mbps digital. The mediumband data may be transmitted realtime or recorded; the wideband data may be transmitted realtime (record capability TBD). Recorded data cannot be transmitted and will only be available at completion of the orbiter mission. The mediumband data may be transmitted via the orbiter S-Band FM link direct to the ground. Wideband data may only be transmitted via the orbiter TDRSS Ku-band relay link and requires TBD timing and synchronization signals from EOS. All data transfer must be hardwire via an interface between the EOS umbilical and the orbiter signal interface panel at Station 576.

Since the above capabilities limit the amount of data that can be obtained to less than that generated by the EOS instruments, each mission will have to be evaluated to determine the interface desired. In the case of EOS-A, an interface will be established which will permit switching the baseband LCU output of the wideband module to the shuttle. This provides an output limited to about 30 Mbps, and can be handled by the orbiter wideband system. Also, a switchable output will be provided for the DCS or mediumband data (OBC memory dump, NBTR playback, or sensor) handled by the modulation processor in the C&DH module. Both interfaces (mediumband and wideband) are made through the spacecraft umbilical.

3.5 CAUTION AND WARNING

Caution and warning capability is provided in the orbiter for payloads to alert the crew to anomalies which require flight crew attention. Up to 50 parameters (analog and digital) may be monitored. These signals will be monitored by sensors hardwired to conditioning circuitry in the EOS Signal Conditioning and Control Module (SCCM). The SCCM, in turn, is directly wired to the orbiter through the spacecraft umbilical which provides the necessary sensor biasing power. The orbiter compares these inputs against predetermined limits and activates warning devices to the crew. Backup caution and warning signals are obtained via the performance monitoring of selected functions in the narrowband telemetry data as defined in Paragraph 3.3.2.

The crew may respond to caution and warning signals via on-board generated commands (see Paragraph 3.3.1) or via special safing commands. These commands are low-level (+5 VDC) and high level (+28 VDC) discrete signals applied directly to logic or relays in the EOS SCCM for control of the anomalous functions. The signals are transferred through the spacecraft umbilical under orbiter software control. These safing signals are also used to provide deactivation of pyro and solenoid devices used for deployment of EOS appendages. A fail-safe design will be used which assures activation of these devices after demating of the umbilical.

All caution and warning and safing command signals must be decided on a mission to mission basis. Further study of applicable functions needs to be performed as mission requirements develop. A preliminary listing of caution and warning functions requiring monitoring is presented in Table 4-1 of Section 4, Payload Shuttle Operation.

3.6 RF COMMUNICATIONS

RF communication capability to a detached payload is limited to forward command capability of 8 kbps (see Paragraph 3.3.1) and 16 kbps of data retrieval capability (see Paragraph 3.3.2). This does not provide any advantage over direct communication with EOS from STDN or TDRSS. The orbiter RF link will be compatible with the STDN transponder on EOS and can be used as a backup to direct command and data transmission to STDN or TDRSS during retrieval operations.

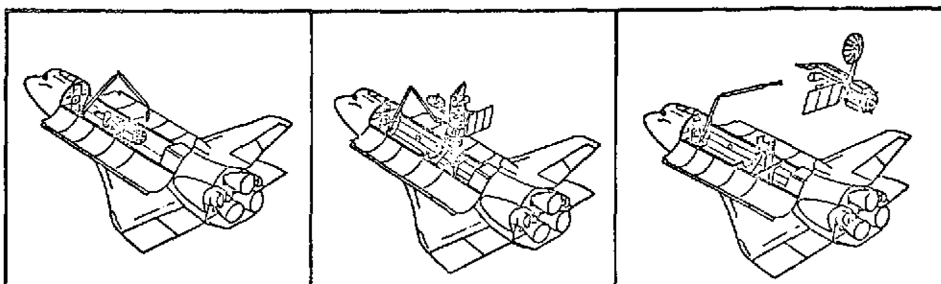
3.7 RESUPPLY

There are no particular unique electrical interface problems associated with a resupply mission. It will require disabling of the EOS power bus during a module exchange maneuver, but this is available as a result of current umbilical power interface requirements. A continuity loop through all module/structure interface connectors will be incorporated as part of the SCCM to provide an indication of proper electrical mate. Also, electrical connector functions will be combined to minimize the number interface connectors required.

3.8 SUMMARY

There are no basic incompatibilities in providing an electrical interface between the shuttle orbiter and the EOS spacecraft. In general, all orbiter/EOS operations will be limited to a hardwire interface while the EOS is stowed or positioned for deployment. The detached mode of operation will be RF and used only as a backup to the standard EOS STDN/TDRSS communications links.

The electrical interfaces are summarized in Figure 3-6. Note that the interface is identical for both the stowed and positioned configurations. All of the attached interfaces are handled through the spacecraft umbilical connector. The power interface will be handled through Station 695 of the orbiter; all other interfaces will be handled through the signal and control utilities payload interfaces in the shuttle bay forward bulkhead at Station 576 (see Figure 3-7). Figure 3-8 shows a schematic of the hardwire interfaces. Holding relays, which are activated by shuttle power, are used to protect the EOS spacecraft against possible shorts in the umbilical interface. These relays automatically open when the EOS/orbiter interface connector is demated.



INTERFACE	STOWED (HARDWIRE)	POSITIONED (HARDWIRE)	DETACHED (RF)
POWER	REGULATION	REGULATION	EOS INTERNAL
COMMAND	8 Kbps (2Kbps EOS DATA) OBC DECOM	8Kbps (2Kbps EOS DATA) OBC DECOM	PRIME - STDN OR TDRSS BACKUP - SHUTTLE 8 Kbps
TELEMETRY	16 Kbps OBC FORMAT (TBD)	16 Kbps OBC FORMAT (TBD)	PRIME - STDN OR TDRSS BACKUP-SHUTTLE 16 Kbps
DATA	MOD. PROC. 1.024 Mbps MB LCU LINK 50 Mbps WB	MOD. PROC. 1.024 Mbps MB LCU LINK - 50 Mbps WB	PRIME - STDN OR TDRSS/FULL CAPABILITY BACKUP-NONE
C&W	SHUTTLE DISPLAY (35)	SHUTTLE DISPLAY (35)	NONE

FIGURE 3-6 SUMMARY OF ELECTRICAL INTERFACES

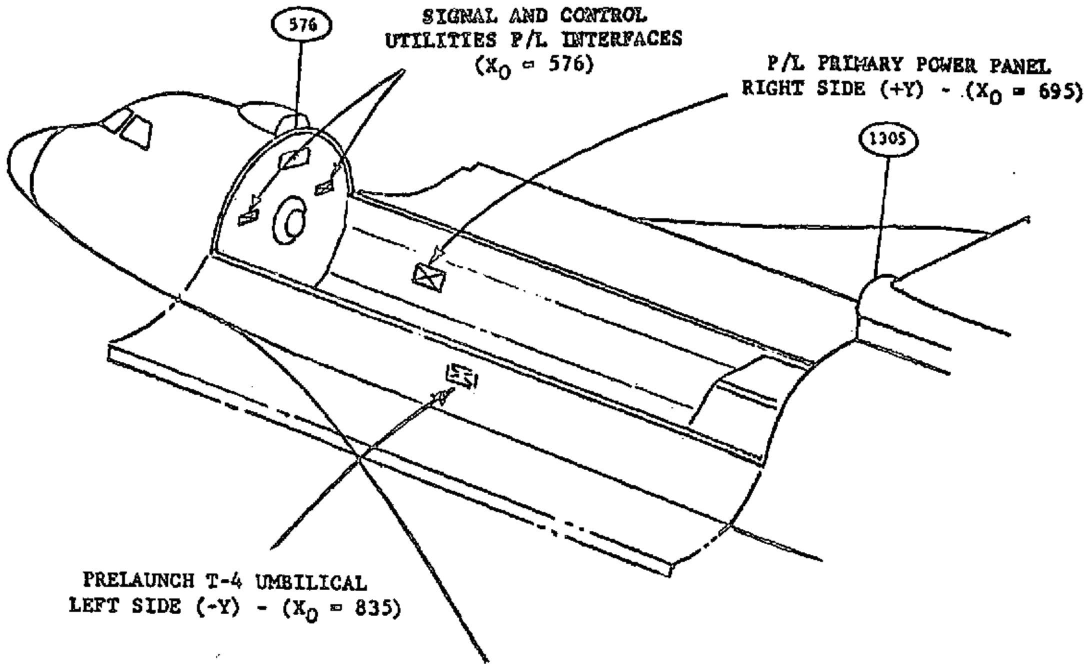


FIGURE 3-7 EOS/ORBITER ATTACHED INTERFACE

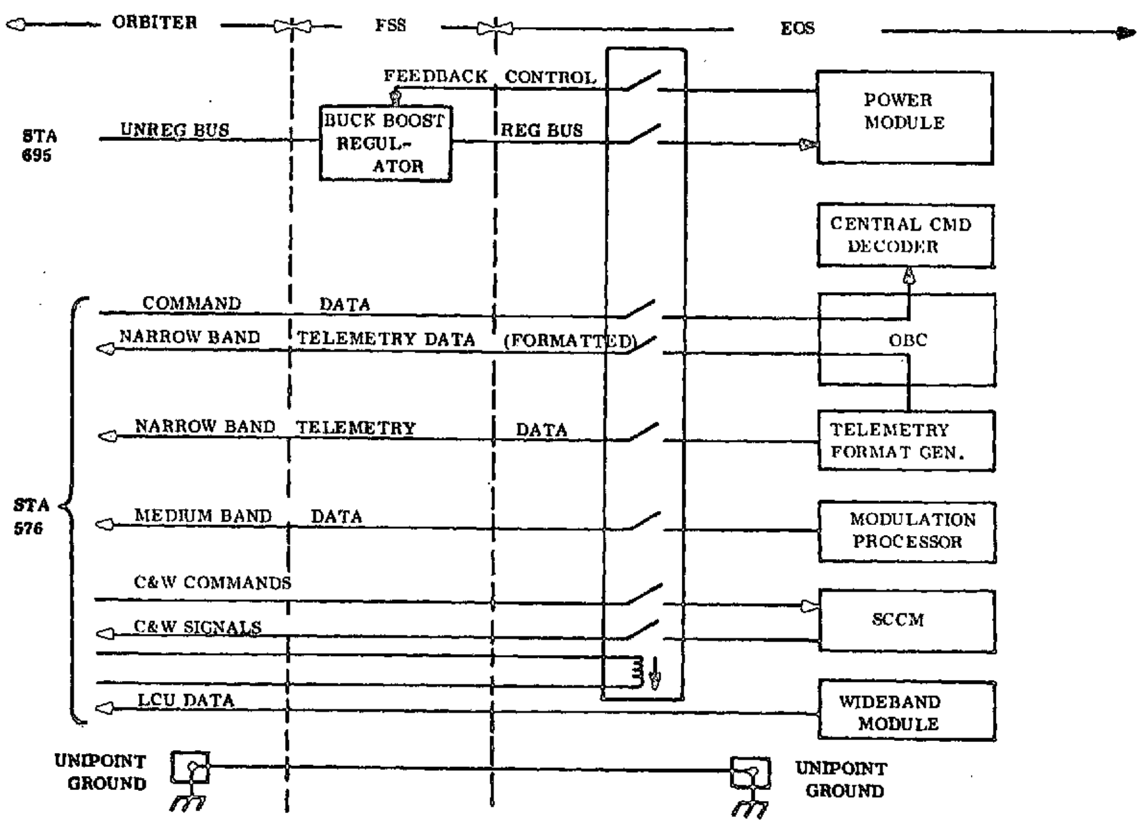


FIGURE 3-8 EOS/ORBITER ELECTRICAL INTERFACE (HARDWIRE)

The orbiter payload support capabilities discussed in this section were based on handling of a single payload (EOS). If multiple payloads are considered, some of the capabilities will have to be shared. When the payloads that will share the cargo bay with EOS are defined a separate evaluation will have to be made to confirm that the shuttle electrical support capability is not exceeded.

SECTION 4.0

PAYLOAD SHUTTLE OPERATIONS

4.1 INTRODUCTION

This section describes the compatibility between EOS and Shuttle operations. It also develops preliminary requirements which the EOS placed on the Shuttle system for operational support. In general, the Shuttle is effective in supporting the operations of the EOS during all phases of the mission. However, in some cases, notably during prelaunch operations, the "preferred way of doing business" requires modification to fit the overriding Shuttle operational flow.

4.2 PRELAUNCH MATE AND SERVICING

Current ground flows for the Shuttle system indicate that loading the payload in the Orbiter cargo bay will be initiated at approximately L-91 and continue for 4 hours. This operation takes place in an Orbiter Processing Facility (OPF) as do all other EOS/ Shuttle activities prior to moving to the launch pad. The EOS/Shuttle integration timeline is illustrated in Figure 4-1. Installation of the payload is followed by verification of the Orbiter/EOS interface connections and final closeout of the payload at approximately L-69. This latter event is critical since it severely restricts access to the spacecraft for almost three days prior to launch, a vastly different situation than is currently practiced with expendable launch vehicles.

After completion of Orbiter processing, it is moved to the Vehicle Assembly Building (VAB) for mating with other elements of the Shuttle which is then moved to the launch pad. After the Shuttle is mated to the pad, access to the EOS may be obtained via a payload changeout room or the Station 576 crew hatch (approximately 4 hours beginning at L-10). Although this capability is provided, the need for physical access to the EOS during this time is not currently anticipated. A potential future requirement may arise if a cryogenic cooling system is used to support advanced sensors. Except for this possibility, all physical servicing of the EOS, including propellant tank loading and pyrotechnic device installation (at L-73), will be completed prior to the spacecraft installation in the cargo bay.

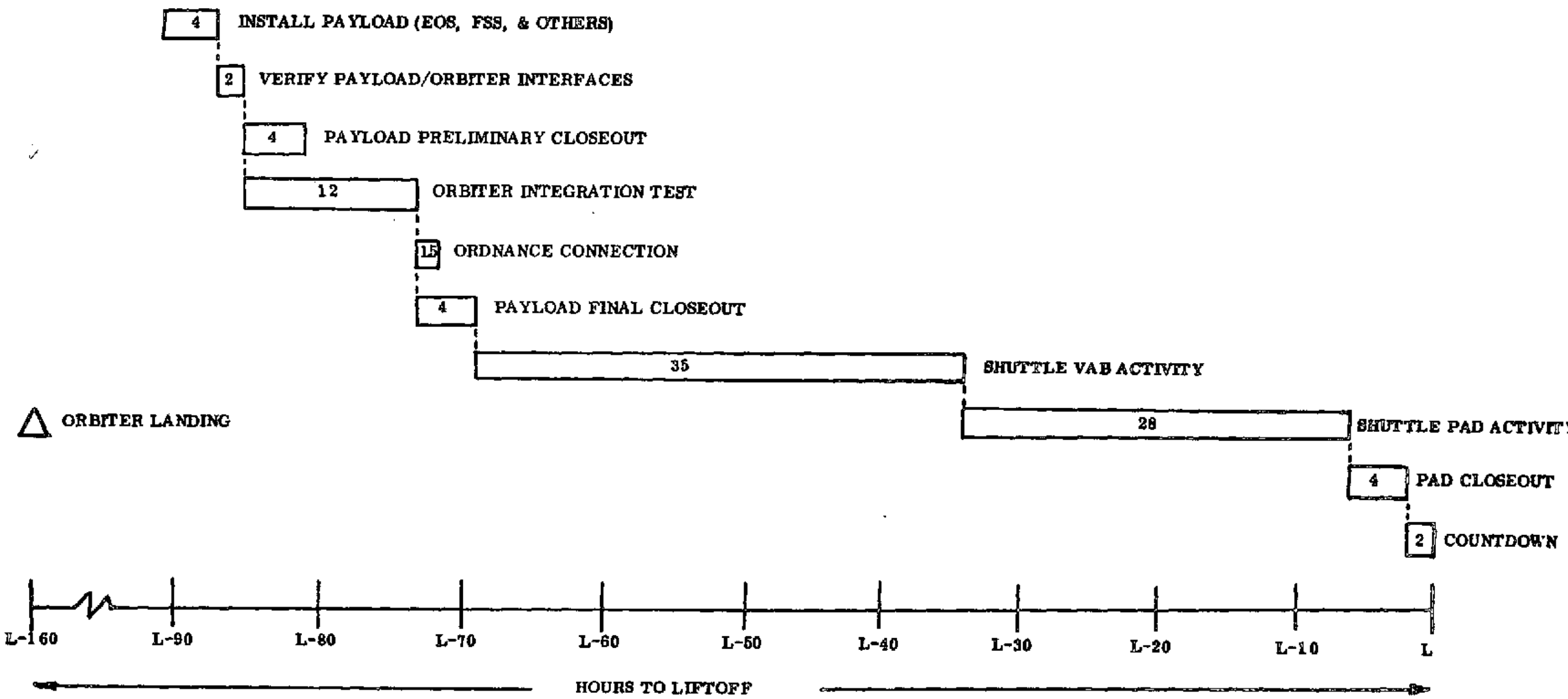


FIGURE 4-1 EOS/SHUTTLE INTERFACE GROUND FLOW

4.2.1 SATELLITE CHECKOUT

Complete satellite systems tests will be conducted at the "factory" level. After a routine incoming inspection at WTR, the critical portions of the complete factory test sequence will be replicated. These will be completed before Orbiter installation which begins at approximately L-91. Checkout of the EOS on the ground following this activity will be restricted to critical measurement monitoring and caution/warning monitoring. Electrical power for these spacecraft functions as well as all others through the Ascent Phase will be supplied by GSE or the Orbiter.

Once the total Shuttle/Payload system has been mated to the launch pad, comprehensive limit checking of EOS subsystems will again be performed at the module and submodule level (beginning at L-10). This activity will be conducted with EGSE via the Station 835 prelaunch umbilical. Since all of the checkout and servicing activities will be completed well before launch, no connection via the T-0 launch umbilical is anticipated. Following removal of the prelaunch umbilical, all monitoring of EOS status will be performed via the Orbiter's interleaved telemetry bit stream.

4.2.2 EOS/ORBITER INTERFACE VERIFICATION

Installation of the EOS into the Orbiter cargo bay will be accompanied by a comprehensive verification of all electrical interfaces. As suggested by RI, the EOS and FSS will be mated and structural interfaces verified prior to L-91. This sequence is expected to take 36 hours as shown in Figure 4-2.

Verification of the EOS/Orbiter electrical interfaces will be accomplished during the two-hour interval beginning at L-87. As currently planned, these interfaces will consist of three functional sets of connections: one set to the Station 835 prelaunch umbilical for EOS checkout and monitoring by EGSE, a second set to the Station 576 payload utility panels for caution/warning, command, and performance monitoring by the Orbiter avionics, and the third set to the right-hand sidewall power panel (Sta.695) for electrical power (see Figure 4-3).

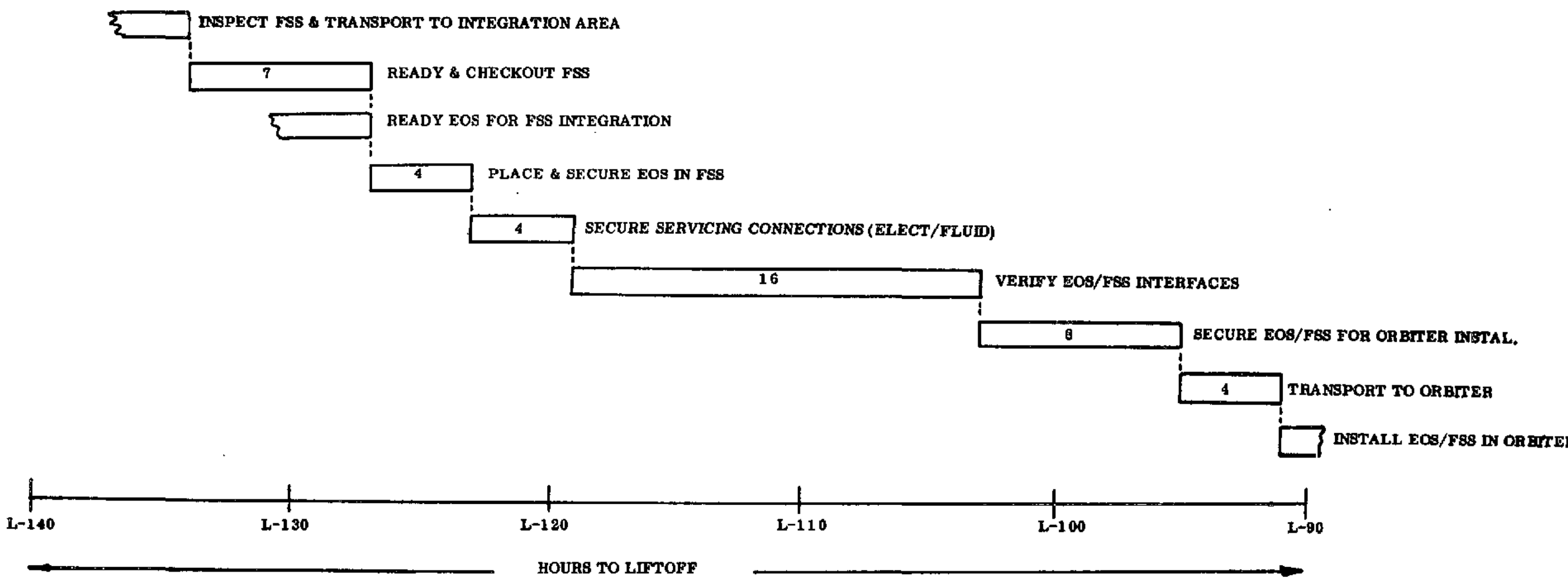


FIGURE 4-2 EOS/ FSS INTEGRATION FLOW

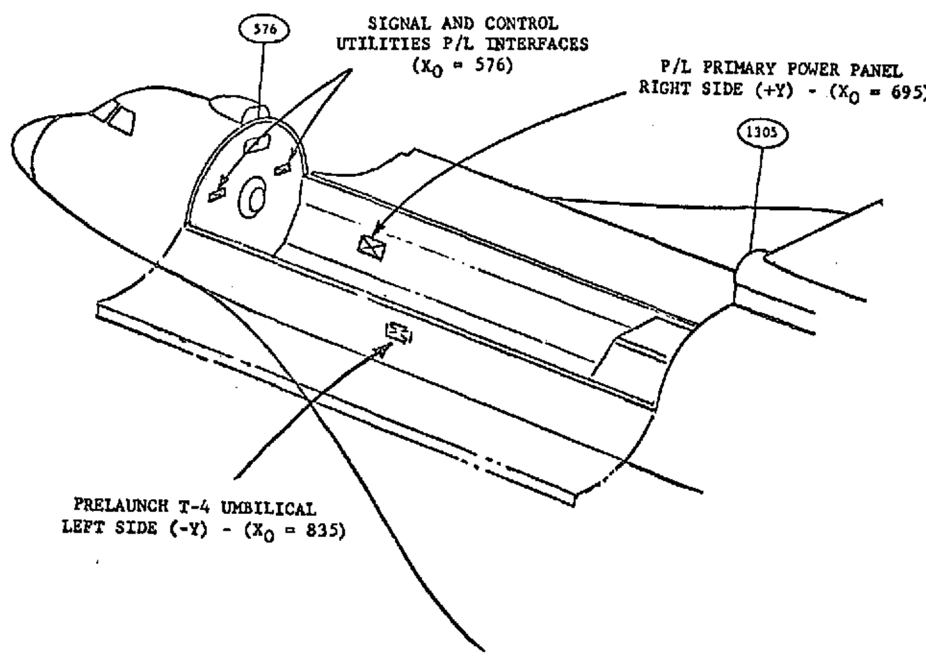


FIGURE 4-3 PAYLOAD/ORBITER ELECTRICAL INTERFACES

No fluid connection interfaces with the Orbiter are anticipated at this time. Although a potential requirement to provide emergency propellant dump lines has been considered such a capability is felt to be unnecessary. This decision is based on discussions with NASA personnel at JSC and Headquarters and is discussed in Appendix B.

4.3 DATA MONITORING DURING ASCENT

During the Ascent Phase, low bit rate data on critical subsystems will be interleaved with the primary telemetry of the Orbiter. In addition, caution and warning status data will continue to be hardwired to the Orbiter, probably to the Mission Specialist Station. Although analysis of EOS systems has not progressed to the point where detail identification of the critical data points is possible, generic functions can be identified. Table 4-1 contains a list of potential functions to be monitored. It also notes which of these are likely to be considered as candidates for caution and warning status. All of the data signals and the return command link to the spacecraft will be carried via hardwire cable to the interface, and located at the Station 576 bulkhead.

TABLE 4-1 GENERIC SUBSYSTEM FUNCTION MONITORING

SUBSYSTEM	FUNCTION MONITORED	CAUTION/WARNING
ATTITUDE CONTROL	REACTION WHEEL STATUS ELECTRONICS TEMPERATURE PNEUMATICS TEMP & PRESSURE	DOUBTFUL DOUBTFUL POSSIBLE (PRESSURE)
ORBIT ADJUST	PROPELLANT TANK & PLUMBING PRESSURE & TEMP THRUSTER VALVES STATUS	PRIME CANDIDATE (PRESSURE) PROBABLE CAUTION
MECHANICAL	SQUIB ARM POWER STATUS TIMER STATUS LATCH STATUS FOR MODULES/DEPLOYABLES	PROBABLE POSSIBLE PROBABLE
THERMAL	SUBSYSTEM TEMPERATURE	POSSIBLE FOR FEW
POWER	BATTERY TEMP BATTERY VOLTAGE LOAD CIRCUIT SWITCH STATUS UMBILICAL CONNECT STATUS	PROBABLE PROBABLE POSSIBLE FOR FEW POSSIBLE
TT&C	COMMAND TRANSLATOR STATUS CLOCK ERROR RATE ANTENNA DRIVE TEMPERATURE	DOUBTFUL DOUBTFUL DOUBTFUL
DATA PROCESSING	CENTRAL PROCESSOR ERROR	DOUBTFUL

As noted above, these in-flight service panels provide the prime interface for all electrical data signals and the right-hand side power panel (Sta.695) will be used for electrical power transfer from the Orbiter. A problem with both of these interfaces is the requirement to break and remate them. The EOS/Shuttle cost analysis (see Section 5.0) indicates that on-orbit servicity of the spacecraft is cost effective. This requires that electrical power and data interface connections be remade after docking. Alternate methods for accomplishing this function are defined in paragraph 4.5.

4.4 ORBITAL OPERATIONS

The compatibility of the EOS and Shuttle after insertion into low earth orbit can be discussed in terms of four phases: post-insertion, pre-separation checkout, separation, and post-separation. The first of these phases is virtually identical to the ascent to low earth orbit. After attainment of orbit and opening of the cargo bay doors, the prime activity of the Orbiter will concern status checking and navigation updating. It is expected that EOS monitoring will remain at the same level and not be interrupted by this Orbiter activity.

After completion of Orbiter-required activities a pre-separation checkout of the EOS will be conducted via the hardwire interface with the Orbiter avionics. The primary purpose of this checkout activity is to assure that the EOS may be safely deployed and recovered, if necessary (see Figure 4-4a). The assurance of deployment safety is relatively simple and will involve electrical continuity checks and visual inspection to check the structural integrity of the spacecraft. This means that no damage has been incurred and that there has been no premature full or partial deployment of the spacecraft appendages.

The portion of this checkout which is conducted visually will be carried out in several stages as the vehicle attitude in the bay is changed. All will involve direct visual access via the operator viewing windows in the forward bulkhead and will also utilize the TV monitors placed at various locations in the cargo bay (locations not yet identified by Space Shuttle Project Office). If necessary, the TV camera located on the Shuttle Attached Manipulator System (SAMS) may also be used. The steps in the visual inspection activity are tied to the total deployment and separation sequence to assure full coverage of the vehicle.

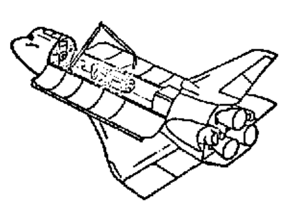
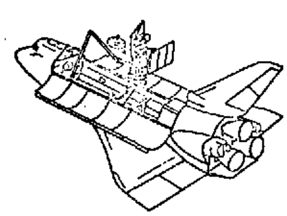
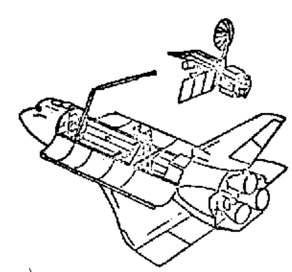
		

FIGURE 4-4a SPACECRAFT IN SHUTTLE RETENTION CRADLE	FIGURE 4-4b SPACECRAFT ATTACHED TO POSITIONING PLATFORM	FIGURE 4-4c SPACECRAFT IN SHUTTLE LOITER MODE
• CAUTION & WARNING MONITORING • STATUS/LIMIT CHECKING OF SUBSYSTEMS & INSTRUMENTS • SPACECRAFT OBP MEMORY UPDATING • PRE-DEPLOYMENT CHECKOUT - HARDWIRE & MECHANICAL INTERFACES - ELECTRICAL CONTINUITY - VISUAL INSPECTION	• CAUTION AND WARNING MONITORING • DEPLOYMENT OF APPENDAGES • STATUS/LIMIT CHECKING AND PRELIMINARY FUNCTIONAL CHECKING OF SUBSYSTEMS AND INSTRUMENTS • SPACECRAFT OBP MEMORY UPDATING • PRE-SEPARATION CHECKOUT - R.F. INTERFACE - ELECTRICAL CONTINUITY - VISUAL INSPECTION • VERIFY RECAPTURE & RETRIEVAL CAPABILITY	• EOS ON SPACECRAFT ACS, POWER AND COMMUNICATIONS • ACTIVATION AND CHECKOUT UNDER GROUND CONTROL • REMAINDER OF SPACECRAFT DEPLOYMENTS (IF REQ'D)

FIGURE 4-4 OPERATIONAL SEQUENCE FOR SPACECRAFT CHECKOUT IN SHUTTLE ORBIT

The checkout routine needed to assure recoverability of the EOS is more complicated than that of checking its "separability" (see Figure 4-4b). The general approach to defining this concept assumes that full activation and checkout of the EOS is best performed with the spacecraft physically separated from the Shuttle. It is therefore necessary to define a simple test sequence which assures minimum operability of the EOS so that it can be recaptured and retrieved by the Orbiter if necessary.

The question of responding to a detected failure after initial deployment is a very complex one. However, regardless of how the problem is resolved (i.e., immediate versus delayed servicing, or return to the ground), any EOS program mode other than expendable requires this type of precautionary pre-separation checkout.

The detailed checkout operations cannot be developed until subsystem design is completed. However, an initial concept is shown in Table 4-2. For each major assembly in the EOS subsystems, the table indicates which is to be activated prior to separation and which is to be tested. A question which has not been resolved concerns the feasibility and desirability of deploying the solar array and testing it in the cargo bay.

The separation activity is carried out after it has been determined that the EOS can successfully survive alone and is capable of being retrieved should a later contingency occur. The entire deployment sequence implies that various inhibit signals be present (i.e., to prevent normal ACS operation). Once safely released, the Orbiter will move off to a safe distance and the spacecraft can be fully activated via its S-band uplink from the ground (see Figure 4-4c).

At this time, a thorough vehicle activation and checkout will be conducted under ground control via direct RF(*). This can be carried out using either the S-band or TDRSS

(*)This activity may also be conducted under remote control of the Orbiter via an RF link. It is not felt, however, that the magnitude of this extended activation and checkout is within the scope of the Orbiter avionics capability. Another option which may be desirable for selected EOS missions is to conduct this activity under ground control, but to utilize the Orbiter avionics to relay engineering data and commands between the EOS and the ground. This option becomes especially desirable for those spacecraft configurations which do not include a TDRSS link.

TABLE 4-2 PRE-SEPARATION ACTIVATION AND CHECKOUT OF EOS

SUBSYSTEM/COMPONENT	ACTIVATE	CHECKOUT
Attitude Control		
Backup Controller	x	x
Magnetic Torquers	x	x
Momentum Wheel	x	
Wheel Electronics	x	x
Star Tracker		
IRU Platform	x	
Solar Aspect Sensor	x	x
Remote TLM and CMD	x	x
Power and Solar Array		
Central Control Unit	x	x
Power Regulator	x	x
Power Control Unit	x	x
Battery	x	x
Remote Decoder MUX	x	x
Solar Array		
Solar Array Drive		
Array Shunt Panel		
Communications and Data Handling		
S-band Transponder	x	x
Modulation Processor	x	x
Central Command Decoder	x	x
Format Generator	x	x
Clock	x	x
Remote Decoder/MUX	x	x
Data Collection Subsystem		
Tape Recorder		
Computer	x	x
S-band Antenna		
TDRSS Transponder		
TDRSS Antenna		

SUBSYSTEM/COMPONENT	ACTIVATE	CHECKOUT
Thermal		
Thermal Coatings	N/A	N/A
Heaters	x	x
Insulation Blankets	N/A	N/A
ACS Thermal Control	x	x
C&DH Thermal Control	x	x
Power Thermal Control	x	x
WB Thermal Control		
Prop Thermal Control	x	x
Propulsion		
Pneumatics		
Orbit Adjust	(Not Required For	
Orbit Transfer	Subsystem)	
Wideband		
Multiplexer		
MUX Encoder	(Not Required For	
QPSK X-band Mod	Subsystem)	
PCM-FM Mod		
TWT/Power Supply		
Elec. Gimbal System		
Antenna and Support		
Tape Recorders		
Remote TLM and CMD		
Harness and Signal Conditioning		
Harness - ACS	x	x
Harness - Power	x	x
Harness - C&DH	x	x
Harness - WB		
Harness - VEH	x	x
Signal Conditioning	x	x
Remote TLM and CMD	x	x

(K-band) up- and downlinks. Unless some unusual contingency arises, the Orbiter is not expected to be directly involved in these operations, although it may be standing by. If some antenna or array deployment has been left for this post-separation phase, however, it may be desirable for Orbiter-located TV cameras to monitor these events.

In the same manner, the Shuttle Orbiter will not nominally be involved with the transfer of the EOS to its mission orbit. Spacecraft stabilization, orientation and initiation of the orbit transfer function will all be done under ground control. If desired and feasible from a safety point-of-view, the Orbiter may take a stand-off position and use its TV cameras to monitor the portion of the sequence conducted at the parking orbit.

4.5 SPACECRAFT RETRIEVAL

Retrieval of the EOS by the Orbiter may be accomplished for either of two reasons: stowage in the cargo bay for return to the ground, or manipulation for on-orbit resupply. Up to the time of capture by the SAMS, the operations are the same for both goals.

The EOS, acting under ground control, will cease its nominal mission functions and return to the Orbiter vicinity (approx. 250 n.mi.). It is assumed that final rendezvous and docking will be performed actively by the Orbiter with the EOS cooperative. Before closing with the EOS, however, the Orbiter must be totally assured of the safe nature of the EOS. Thus there is a requirement for a pre-docking checkout of the EOS to assure that all safety parameters are in an acceptable range and that there are no configurational hazards (e.g., TDRSS antenna is stowed). This checkout may be conducted by the EOS mission control center via a direct spacecraft/ground link or may be performed remotely by the Orbiter. The selection of the optimum technique will depend in large measure on the capability of the Orbiter to conduct such a test and the available link time to a STDN or TDRSS station.

After the pre-docking safety checks have been made, the EOS will be recaptured by the Orbiter with the SAMS and either placed in the FSS retention cradle or docked to the positioning platform. (It is assumed that the FSS elements and Orbiter interfaces have been checked out in advance of this event.) The EOS/Orbiter electrical signal and power interfaces will be re-mated and, after verification of these interfaces, the EOS may be deactivated to its desirable state.

The reestablishing of the electrical interfaces is required functionally, but has not been investigated mechanically. The in-flight mating of electrical connectors may be performed with the use of SAMS or through an as-yet unidentified capability of the FSS. Another alternative approach is to use the EVA capability of the Shuttle Orbiter crew. The necessary trade-study which would identify the most cost-effective approach, is beyond the scope of the current study.

SECTION 5.0

SHUTTLE MODE COST ANALYSIS

5.1 <u>INTRODUCTION & SUMMARY</u>

5.1.1 INTRODUCTION

The analysis to determine the most cost effective method of using shuttle for the EOS program is the key trade-off addressed in this report. The tradeoff involves evaluating the cost impacts of using shuttle:

1) As a launch vehicle
   (expendable spacecraft concept)
2) To deliver and return the spacecraft
   (ground serviceable spacecraft concept)
3) To deliver and service the spacecraft
   (on-orbit serviceable spacecraft concept)
4) To perform a combination of the above functions

5.1.2 ANALYSIS APPROACH

The analysis of the cost impacts of these alternate modes of using shuttle was approached by establishing the following tasks.

1) Establish a simplified mission model and orbits compatible with the EOS program definition
2) Establish estimates and assumptions on the spacecraft (lifetime, weights, costs), the shuttle (costs, support requirements and weights), ground servicing and the alternate on-orbit servicing concepts.
3) Perform a cost analysis of the alternate shuttle modes for the nominally assumed values established in the proceeding task. This analysis assumes no spacecraft failures and is performed for the following modes
   - expendable spacecraft concept
   - ground servicable spacecraft concept
   - combined orbital and ground servicable spacecraft concept
   - orbital servicable only spacecraft concept

4) Establish a reasonable range for the variables developed in Task 2 particularly where the data is "soft" or the results may be extra sensitive to the assumed values. (i.e., refurbishment costs, launch costs, number of spacecraft failures)
5) Evaluate the impact of the range of variables selected and determine which variables significantly impact the study results.
6) Verify the assumptions for the "sensitive" variables.
7) Prepare recommendations on the cost effective use of shuttle for the EOS program

5.1.3 SUMMARY

The results of these analyses are as follows:

- The expendable spacecraft mode is the least cost effective (highest cost) of all the cases considered
- On-orbit serviced spacecraft are lowest cost for all cases considered
- Ground serviced and combined ground and on-orbit serviced spacecraft costs are higher (by usually less than 30%) than on-orbit services spacecraft
- On-orbit serviced spacecraft are most cost effective when spacecraft failures are considered

5.1.4 RECOMMENDATION

The most cost effective use of shuttle is achieved by using shuttle to deliver the spacecraft and also assist in servicing the spacecraft to extend its orbital lifetime. The shuttle launched EOS spacecraft should be designed for on-orbit servicing while the spacecraft launched prior to shuttle availability can be designed for shuttle retrieval and ground servicing without incurring significant cost penalties over on-orbit servicing. As the designs of EOS and shuttle mature the shuttle mode analysis can be refined to establish the most cost effective use of shuttle which may include combined on-orbit and ground servicing or may be limited to on-orbit servicing of the spacecraft.

5.2 COSTING CRITERIA AND ASSUMPTIONS

An assessment of the relative merits of the alternate methods of using the shuttle requires that costing criteria and assumptions be established to define the essential differences between the approaches. A nominal set of assumptions were originally established to allow a "first-cut" analysis and determine cost trends. The key assumptions were then varied to establish the sensitivity of the results to these key assumptions. This method of analysis was selected to allow cost trend data to be developed without being overly constrained by the original costing criteria and assumptions.

This section of the report discusses the nominal set of assumptions that were generated to initiate the analysis in addition to defining the selected range of variables used during the sensitivity analysis.

5.2.1 MISSION MODEL & ORBIT

The present definition of EOS includes two similar spacecraft in orbit simultaneously. For the purposes of this analysis a program has been assumed having two spacecraft in orbit at one time over a 10 year program. It has also been assumed that the entire program falls in the shuttle era. That is, the effects of starting with a conventional launch vehicle for the first missions and then transitioning to shuttle were excluded since it was concluded this would complicate the analysis, but not affect the results. The shuttle delivery, retrieval and service orbit has been assumed to be 465 km (250 nm) circular (see Appendix A for discussion of the rational) and the mission orbit has been assumed as 775 km (418 nm). The mission orbit was used to determine the propellant weight and cost required to transfer the spacecraft from the shuttle delivery orbit to the mission orbit and return to the shuttle orbit.

5.2.2 SPACECRAFT COSTS, WEIGHTS & LIFETIME

A basic requirement of this analysis is the availability of non-recurring and recurring costs and weights of the spacecraft under consideration. The costs of the spacecraft and their associated programmatic elements can be expected to vary as a result of their being designed for expend, refurbish or resupply operations. Obviously a returnable spacecraft must be capable of refolding or jettisoning its appendages while an orbital resupplyable spacecraft must provide additional hardware to allow remote disengagement, removal and replacement of modules.

Likewise when spacecraft weights are considered the expendable spacecraft does not require propellant to return it to shuttle while the resupplyable spacecraft weight will increase to allow for handling provisions, resupply latches, electrical disconnects and instrument module structures.

The nominal estimates of the relative costs and weights of these spacecraft options are summarized in Table 5-1. These figures are considered reasonable for this trade study and should be construed as absolute estimates. The cost ratios between the expend mode and the two serviceable modes used slight modifications of the factors actually derived and used in a previous GE study (Payload Utilization of Tug). The refurbishment costs were established using data from AIAA Paper 73-73 in addition to data submitted from vendors and inhouse estimates.

TABLE 5-1
ASSUMED SPACECRAFT WEIGHTS AND COSTS

Mode of Operation	Weight (LBS)		Cost M$
	Delivery	Retrieve	
Expendable S/C	4150	N/A	28.8
Retrievable S/C	4500	3750	30.3
On-Orbit Serviceable S/C	4950	4200	32.8
2 Yr. Nominal Service Mission	3450	2700	6.6
Ground Service	--	--	9.1
On Orbit Service of Failure	1800	1050	3.3

The two year nominal service mission assumes replacement of two subsystems, two experiments, the wideband system, solar array and drive and the propulsion system. The on orbit service of a failure assumes replacement of one subsystem module, one experiment and the propulsion system.

The nominal spacecraft life in orbit has been assumed as 2 years. At the end of 2 years the following actions are taken for each mode of operation.

1) The expendable spacecraft is discarded and replaced with a new spacecraft
2) The ground serviceable spacecraft is replaced, returned to the ground and refurbished for later use.
3) The on-orbit serviceable spacecraft is serviced in orbit. It is discarded at the end of its useful life including servicing.
4) The combined on-orbit and ground serviceable spacecraft is serviced in orbit one or more times and then returned to the ground for ground refurbishment and reuse.

A total lifetime of 10 years has been assumed for a ground serviceable spacecraft while the total lifetime of the on-orbit spacecraft has been varied from 6 years (2 services) to 10 years (4 services) and is discussed further in section 5.2.3.

5.2.3 SHUTTLE COST AND ACCOMMODATIONS

The Shuttle assumptions required for the tradeoff analysis include the shuttle trip charges, requirements for shuttle support equipment including their estimated weights and costs and an establishment of the alternate shuttle on-orbit servicing modes of operation.

The Shuttle cost formula supplied for the EOS study, modified slightly by GE and discussed in more detail in appendix A is:

$$\text{shuttle costs (one way)} = 4.9\text{M} \left[\frac{\text{EOS chargeable Wts}}{21{,}600 \times .78} \right]$$

where -

- Shuttle cargo sharing efficiency of .78 is assumed
- EOS shuttle support systems are only assumed shared with other EOS flights
- Max shuttle one way cost = 4.9M

Shuttle support system definitions for EOS have been established by R.I. and SPAR under separate study contracts to GSFC. These support concepts have been reviewed by GE and the concepts adhered to with some minor modifications which have been coordinated with R.I. These concepts are discussed in section 2 of this report. For this trade off study it has been assumed that the positioning platform will not be required for either the expendable or returnable spacecraft modes. It is assumed that the Shuttle Attached Manipulator System (SAMS) can be used to deploy and retrieve the spacecraft. Unique shuttle equipment required for the on-orbit serviceable mode of operation therefore becomes

1) The Positioning Platform
2) The Special Purpose Manipulator System (SPMS)
3) The Module Exchange Mechanism (MEM)

The assumed weights and costs of this support equipment are summarized in Table 5-2. Non-recurring costs are only shown for the equipment unique to the servicing mission since the other non-recurring costs apply to all missions.

The Shuttle transportation charges can be calculated using the shuttle trip charge formula previously defined, the spacecraft weights defined in table 5-1 and the shuttle support system weights defined in table 5-2. These costs are defined as a function of spacecraft mode of operation and transportation direction in table 5-3.

TABLE 5-2
ASSUMED SHUTTLE SUPPORT SYSTEM WEIGHTS & COSTS

Support Equipment	Weight (LBS)	Cost M$		
		NonRecurring	Recurring	Refurbish
Retention Cradle	600	--	0.3	0.01
Positioning Platform	1300	1.0	0.5	0.04
Data Management, Electrical Power & Thermal Control	--	--	0.5	0.15
Module Magazine & Module Exchange Mechanism	2200	(10.0)*	(2.5)*	0.5

*These costs will not be chargeable to the EOS program

TABLE 5-3
SHUTTLE TRIP CHARGES (LAUNCH & RETRIEVE)

Spacecraft Mode	Trip Direction	S/C Weight (LB)	Shuttle Chargeable Supt. Wt (LB)	Total Weight (LB)	Shuttle Trip Charge (M$)
Expendable Spacecraft	Up	4150	600	4750	1.38
Ground Serviceable S/C	Up	4500	600	5100	1.48
	Down	3750	600	4350	1.26*
On-Orbit Serviceable S/C	Up	4950	600	5550	1.61
	Down	4200	600	4800	1.39*
2 Yr Service Mission	Up	3450	3500	6950	2.01
	Down	2700	3500	6200	1.80*
Failure, Service Mission	Up	1800	3500	5300	1.54
	Down	1050	3500	4550	1.32*

*Charges for the down portion of a shuttle round trip will be costed at no less than the up portion of the round trip

Four alternate on-orbit servicing concepts have been assumed for this tradeoff analysis. The first two concepts involve combined on-orbit and ground servicing while the other two are restricted to on-orbit servicing with the spacecraft discarded two years after the final on-orbit servicing. The concepts studied are:

Combined Ground and On-Orbit Servicing

1) One on-orbit service and then return the spacecraft to the ground for refurbishment and reuse
(this sequence is repeated until 10 years on orbit life is reached and then the spacecraft is discarded)
2) Two on-orbit services and then return the spacecraft to the ground for refurbishment and reuse
(This sequence also assumes a total on-orbit life of 10 years prior to discarding the spacecraft)

On-Orbit Servicing

3) Two on-orbit servicings of the spacecraft and then discard the spacecraft
(This sequence assumes a total of 6 years of on-orbit life)
4) Four on-orbit servicings of the spacecraft and then discard the spacecraft
(This sequence assumes a total on-orbit life of 10 years)

5.2.4 GROUND COSTS

The ground costs for logistics manpower has been assumed to be:

1) Expendable Spacecraft Mode (.1M/yr)
2) Ground Serviceable Spacecraft Mode (.2M/yr)
3) On-Orbit Serviceable Spacecraft Mode (.2M/yr)

It should be noted that these costs do not include any refurbishment costs or the costs of spare hardware which is costed elsewhere.

5.2.5 SELECTED RANGE OF VARIABLES

The nominal assumptions established for the shuttle mode cost analysis have been discussed in sections 5.2.2, 5.2.3 and 5.2.4 of this report. These assumptions have been used to establish nominal costs of the alternate shuttle modes of operation. This section defines the range of values selected for some of the variables to establish sensitivities of the analysis to those variables. These values are summarized in Table 5-4.

TABLE 5-4
RANGE OF VARIABLES SELECTED FOR SENSITIVITY ANALYSIS

VARIABLE	NOMINAL	RANGE
REFURBISHMENT COST GRD REFURBISHMENT (2 YR) GRD REFURBISHMENT (FAILURE) ON-ORBIT SERVICE (2 YR) ON-ORBIT SERVICE (FAILURE)	 9.1M 7.6M 6.6M 3.3M	 9.1M TO 15.1M 7.6M TO 12.6M 6.6M TO 9.8M 3.3M TO 6.6M
LAUNCH COSTS	SHUTTLE TRIP FORMULA	SHUTTLE TRIP FORMULA TO FULL ONE WAY CHARGE OF 4.9M
SPACECRAFT COSTS EXPENDABLE GROUND SERVICEABLE ON-ORBIT SERVICEABLE	 28.8M 30.3M 32.8M	 18.9M TO 37.9M 20.2M TO 40.4M 21.9M TO 43.7M
NUMBER OF FAILURES	ZERO	ZERO TO THREE
COSTS	RECURRING COSTS ONLY	RECURRING COSTS ONLY TO RECURRING PLUS Δ NON-REC. REQ'D FOR SERVICING
GROUND COSTS EXPENDABLE GROUND SERVICEABLE ON-ORBIT SERVICEABLE	 .1M/YR .2M/YR .2M/YR	 .1M/YR .2M/YR TO 2M/YR .2M/YR TO 2M/YR
NUMBER OF SPACECRAFT REQUIRED	TOTAL NO. OF S/C REQ'D FOR 10 YR PROGRAM	S/C REQUIRED FOR 10 YR PROGRAM TO PRORATED COST OF SPACECRAFT FOR 10 YR PORTION OF LONGER PROGRAM

5.3 COST ANALYSIS

The shuttle mode cost analysis has been performed in three stages. The first stage analysis was performed with assumed nominal case variables and no failures. The next stage evaluated the impacts of ranges of variables to establish sensitivities of the analysis to these variables. The third and final stage of the analysis involved further investigation of the most sensitive variables and an analysis of the revised nominal cases using "best estimate" values for the variables while also including a nominal number of failures in the costing. These three stages of the analysis are discussed in the following three sections and are followed by a summary of the cost analysis.

5.3.1 COST ANALYSIS OF NOMINAL CASE (NO FAILURES)

The Nominal Case cost analysis was performed using the values for the variables as defined in sections 5.2.2 to 5.2.4 and for the following modes of operation:

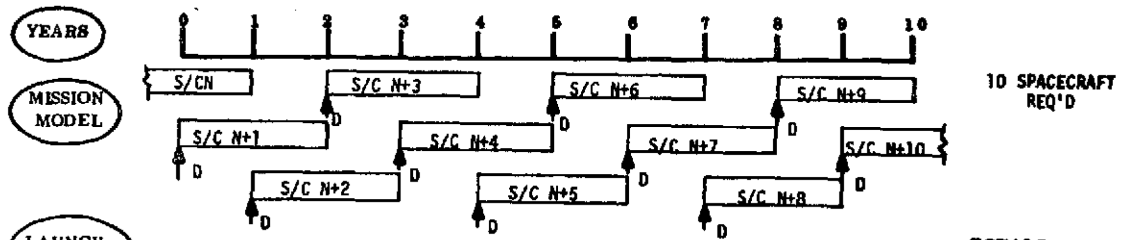
1) expendable spacecraft
2) ground serviceable spacecraft
3) combined orbital and ground serviceable spacecraft
4) orbital serviceable only spacecraft

Each mode of operation was evaluated for two alternate mission models to establish the impact of alternate delivery or other operational concepts.

5.3.1.1 Expendable Spacecraft

The first expendable spacecraft mission model considered is shown in figure 5-1 with launches each year for ten years and spacecraft lives of two years each. Thus 10 spacecraft and 10 launches are required. The launch costs are determined using the shuttle cost formula defined in section 5.2.3 which gives an up trip charge of 1.38 M dollars as defined in table 5-3. The down charge of 1.38 M dollars reflects the criteria that down charges must not be less than up charges. The FSS costs include the recurring costs for a retention cradle (0.3M) and Data Management, Electrical Power, and Thermal Control (0.5M) in addition to the refurbishment cost of the units of 0.16 M. The ground costs are assumed to be 0.1 M/year for a total ground cost of 1.0 M over the 10 year period. There are no SPMS or spacecraft refurbishment costs for the expend mode of operation. The total cost for this mode of operation therefore becomes 319 M dollars for the assumed 10 year program as shown in Figure 5-1.

The alternate expendable spacecraft mode of operation assumes that two spacecraft are launched on one shuttle flight thus requiring only 5 shuttle launches. This method of operation is summarized in Figure 5-2 showing a similar program cost of 319 M dollars. The method of calculating shuttle costs and the assumption that FSS costs for a dual launch will be double the FSS costs for a single launch makes this case identical to the previous case considered. When full launch costs are assumed as one of the sensitivity variables in section 5.3.2 the impact of the alternate methods of operation will be more evident.



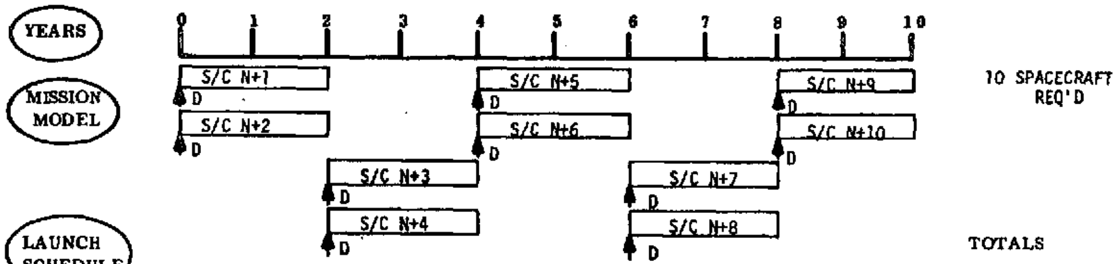
LAUNCH SCHEDULE

											TOTALS
DELIVERY	1	1	1	1	1	1	1	1	1	1	10
SERVICE	-	-	-	-	-	-	-	-	-	-	-
RETRIEVE											

COST ANALYSIS

SPACECRAFT COSTS	28.8 ←	→ 28.8	288
LAUNCH COSTS UP / DWN / TOTAL	1.38 / 1.38 / 2.76 ←	→ 2.76	27.6
FSS COSTS	0.8 0.16 ←	→ 0.16	2.2
SPMS COSTS	N/A		-
GROUND COSTS	0.1 ←	→ 0.1	1.0
REFURBISHMENT COSTS	N/A		-
		TOTAL COSTS	318.8

FIGURE 5-1
COST ANALYSIS - EXPENDABLE SPACECRAFT (SINGLE LAUNCH)



LAUNCH SCHEDULE

											TOTALS
DELIVERY	2	-	2	-	2	-	2	-	2	-	10
SERVICE	-	-	-	-	-	-	-	-	-	-	-
RETRIEVE	-	-	-	-	-	-	-	-	-	-	-

COST ANALYSIS

SPACECRAFT COSTS	57.6	-	57.6	-	57.6	-	57.6	-	57.6	-	288
LAUNCH COSTS UP / DWN / TOTAL	2.76 / 2.76 / 5.52	-	5.52	-	5.52	-	5.52	-	5.52	-	27.6
FSS COSTS	1.6	-	0.32	-	0.32	-	0.32	-	0.32	-	2.2
SPMS COSTS	N/A										-
GROUND COSTS	0.1 ←									→ 0.1	1.0
REFURBISHMENT COSTS	N/A										-
										TOTAL COSTS	318.8

FIGURE 5-2
COST ANALYSIS - EXPENDABLE SPACECRAFT (DUAL LAUNCH)

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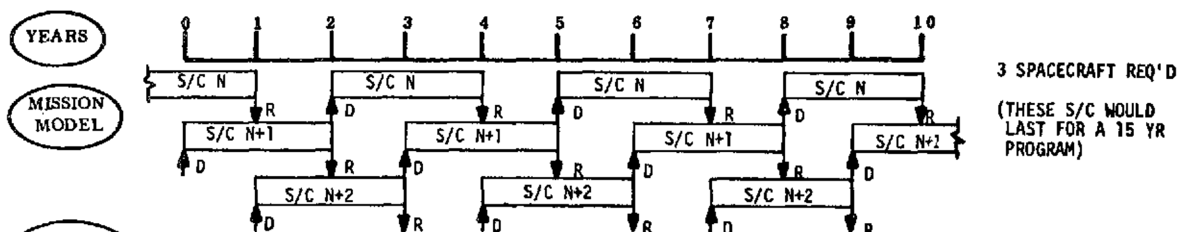
5.3.1.2 Ground Serviceable Spacecraft

Two alternate modes of operating with ground serviceable spacecraft are shown in Figures 5-3 and 5-4.

The first mode assumes launches and retrievals on one year centers with a single spacecraft launched and retrieved on each launch. Three spacecraft are required for this mode of operation giving a total operational life of 15 years. The spacecraft costs on Figure 5-3 reflect the requirement for three spacecraft (no prorating of spacecraft costs) over a ten year period. For a discussion of the impacts of prorating spacecraft costs see section 5.3.2.7.

The launch costs of 29.6 M are established by having five flights with total trip charges of 2.96 M per flight. Spacecraft refurbishment cost of 9.1 M dollars are required in each of the last seven years of the assumed ten year program giving a total refurbishment



LAUNCH SCHEDULE											TOTALS
DELIVERY	1	1	1	1	1	1	1	1	1	1	10
SERVICE	-	-	-	-	-	-	-	-	-	-	-
RETRIEVE	-	1	1	1	1	1	1	1	1	1	9

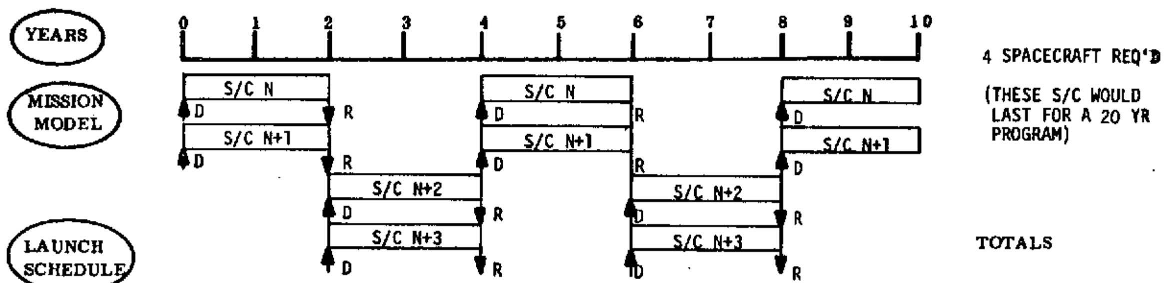
COST ANALYSIS

SPACECRAFT COSTS	30.3 30.3 30.3 - - - - - - -	90.9
LAUNCH COSTS UP / DWN / TOTAL	1.48 / 1.48 / 2.96 ← → 2.96	29.6
FSS COSTS	0.8 0.16 ← → 0.16	2.2
SPMS COSTS	N/A	-
GROUND COSTS	0.2 ← → 0.2	2.0
REFURBISHMENT COSTS	- - - 9.1 ← → 9.1	63.7
	TOTAL COSTS	188.4

FIGURE 5-3
COST ANALYSIS - GROUND SERVICEABLE S/C (SINGLE LAUNCH)

cost of 63.7 dollars. This refurbishment is assumed to return the spacecraft to its original condition and allow two more years of orbital operation prior to the next refurbishment. Total spacecraft orbital life has been limited to ten years. The ground serviceable spacecraft shows a total program cost of 188 M dollars which is a significant cost savings over the expendable spacecraft mode of operation.

The second mode of ground serviceable spacecraft operation shown in Figure 5-4 assumes dual spacecraft launches and retrievals via shuttle. Four spacecraft are required for this mode of operation with a total operational life of twenty years. Here again as in the first ground servicing mode spacecraft costs have not been prorated and show four spacecraft required for the ten year program. The additional spacecraft required for this mode of operation increases the system cost giving a total cost of 210 M dollars.



LAUNCH SCHEDULE

	0	1	2	3	4	5	6	7	8	9	TOTALS
DELIVERY	2	-	2	-	2	-	2	-	2	-	10
SERVICE	-	-	-	-	-	-	-	-	-	-	-
RETRIEVE	-	-	2	-	2	-	2	-	2	-	8

COST ANALYSIS

	0	1	2	3	4	5	6	7	8	9	TOTALS
SPACECRAFT COSTS	60.6	-	60.6	-	-	-	-	-	-	-	121.2
LAUNCH COSTS UP	2.96										
DWN	2.96										
TOTAL	5.92	-	5.92	-	5.92	-	5.92	-	5.92	-	29.6
FSS COSTS	1.6	-	0.32	-	0.32	-	0.32	-	0.32	-	2.9
SPMS COSTS	N/A										-
GROUND COSTS	0.2 ←									→ 0.2	2.0
REFURBISHMENT COSTS	-	-	-	-	18.2	-	18.2	-	18.2	-	54.6
TOTAL COSTS											210.3

FIGURE 5-4
COST ANALYSIS - GROUND SERVICEABLE S/C (DUAL LAUNCH)

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5.3.1.3 Combined Orbital and Ground Serviceable Spacecraft

The primary difference between the two combined orbital and ground serviceable spacecraft concepts is the number of orbital servicings allowed prior to returning the spacecraft to ground for a more complete ground refurbishment. The concept shown in Figure 5-5 has one orbital service prior to retrieval while the other allows two orbital services prior to retrieval. In each case three spacecraft are required giving a total life of 15 years and a spacecraft cost of 98.4 M dollars.

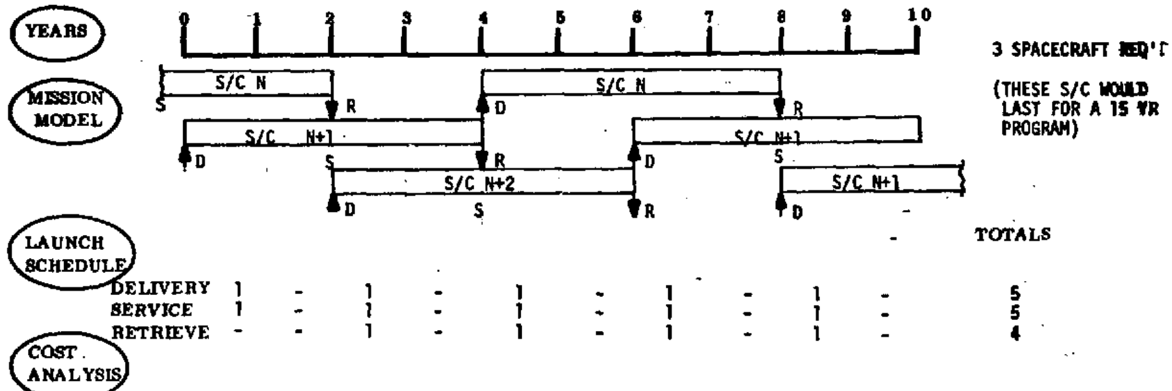
The launch schedule on Figure 5-5 shows one delivery and service mission and four delivery, service and retrieval missions for shuttle. The costs are established from Table 5-3 as follows giving total launch costs of 36.2 M dollars when the shuttle return criteria is incorporated.

• Delivery & Service Mission

Serviceable spacecraft up	= 1.61	3.62	
On-Orbit servicing unit up	= 2.01		
On-Orbit servicing unit down	= 1.80	3.62	(down costs must not be less than up costs)
Delivery FSS Unit down	= 0.17		
		7.24 M	

• Delivery, Service & Retrieve Mission

Serviceable spacecraft up	= 1.61	3.62	
On-Orbit servicing unit up	= 2.01		
On-Orbit servicing unit down	= 1.80	3.62	(down costs must not be less than up costs)
Serviceable spacecraft down	= 1.39		
		7.24 M	



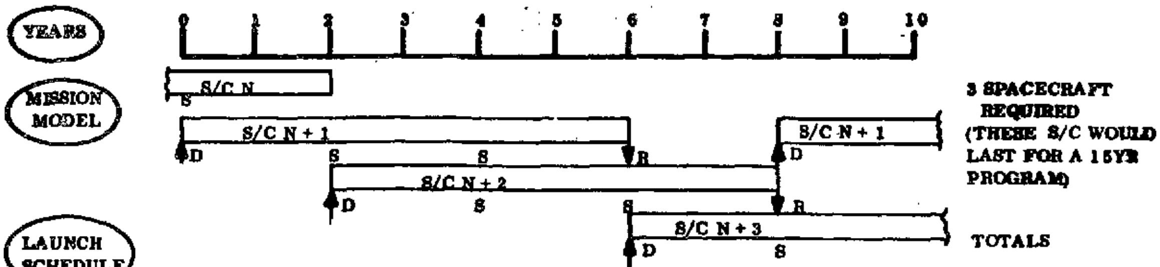
SPACECRAFT COSTS	32.8	-	32.8	-	32.8	-	-	-	-	-	98.4
LAUNCH COSTS UP / DWN / TOTAL	3.62 / 3.62 / 7.24	-	7.24	-	7.24	-	7.24	-	7.24	-	36.2
FSS COSTS	1.3	-	.2	-	.2	-	.2	-	.2	-	2.1
SPMS COSTS	-	-	0.5	-	0.5	-	0.5	-	0.5	-	2.0
GROUND COSTS	0.2 ←									→ 0.2	2.0
REFURBISHMENT COSTS	6.6	-	6.6	-	6.6	-	15.7	-	15.7	-	51.2
									TOTAL COSTS		191.9

FIGURE 5-5
COST ANALYSIS - COMBINED ON-ORBIT & GRD. SERV. S/C (1 SERV. & RET.)

The FSS costs include retention cradle, positioning platform, and data management, electrical power and thermal control with a total recurring cost of 1.3 M and refurbishment costs of 0.2 M. The SPMS is required for servicing missions at a recurring cost of 2.5 M (which is not chargeable to the EOS Program) and refurbishment costs of 0.5 M. The SPMS includes the module magazine and the module exchange mechanism.

The refurbishment costs vary from 6.6 M for on-orbit servicing to 9.1 M for ground refurbishment giving a total program refurbishment cost of 51.2 M dollars. The combined cost for the spacecraft limited to one on-orbit servicing prior to retrieval is 192 M dollars or slightly more than the ground serviceable spacecraft.

When two on-orbit servicings are assumed prior to retrieval the program costs are reduced to 189 M dollars as shown in Figure 5-6. This cost reduction is due to the reduced number of ground refurbishments required in this mode of operation.



											TOTALS
LAUNCH SCHEDULE											
DELIVERY	1	-	1	-	-	-	1	-	1	-	4
SERVICE	1	-	1	-	2	-	1	-	1	-	6
RETRIEVE	-	-	-	-	-	-	1	-	1	-	2
COST ANALYSIS											
SPACECRAFT COSTS	32.8	-	32.8	-	-	-	32.8	-	-	-	98.4
LAUNCH COSTS UP	3.62				3.31						
DWN	3.62				3.31						
TOTAL	7.24	-	7.24	-	6.62	-	7.24	-	7.24	-	35.6
FBS COSTS	1.30	-	.20	-	.19	-	.20	-	.20	-	2.1
SPMS COSTS	-	-	0.5	-	0.5	-	0.5	-	0.5	-	2.0
GROUND COSTS	.20 ←									→ .20	2.0
REFURBISHMENT COSTS	6.6	-	6.6	-	13.2	-	6.6	-	15.7	-	48.7
										TOTAL COSTS	188.8

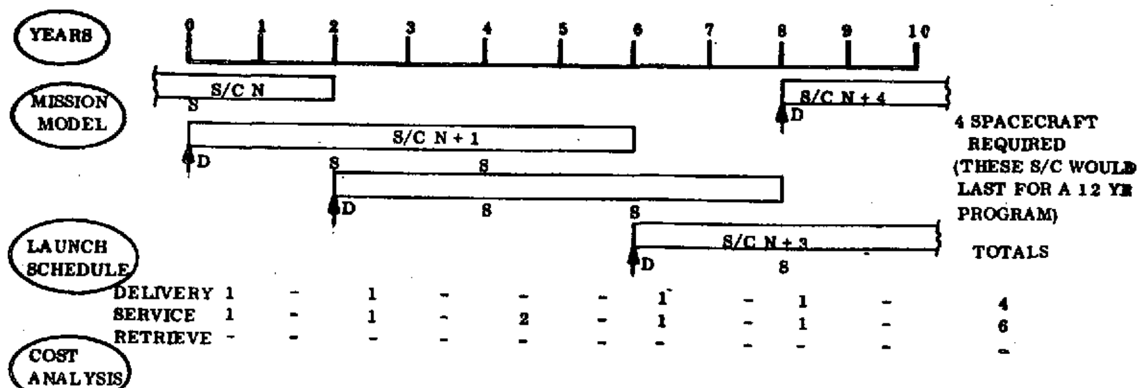
FIGURE 5-6
COST ANALYSIS - COMBINED ON-ORBIT & GRD. SERV. S/C (2SERV. & RET.)

5.3.1.4 Orbital Serviceable Spacecraft

Two orbital serviceable spacecraft options have been considered in the nominal cost analysis. The first assumes that the spacecraft can be serviced twice prior to discarding the spacecraft while the second assumes that a total on-orbit spacecraft life of 10 years can be obtained by servicing the spacecraft four times in orbit.

The first option which is shown in Figure 5.7 indicates that 4 spacecraft are required over a ten year period giving a total spacecraft cost of 131.2 M dollars.

The assumption that the spacecraft is discarded after 6 years in orbit causes the spacecraft costs to increase over the other serviceable concepts considered. The launch costs for this mode of operation are 35.6 M dollars. The refurbishment costs are reduced for this mode since all refurbishment is done in orbit. The cost savings in the launch and refurbishment areas do not however offset the increased spacecraft costs involved in discarding the spacecraft after six years in orbit. This option is therefore the most costly serviceable spacecraft option considered at a program cost of 213 M dollars.

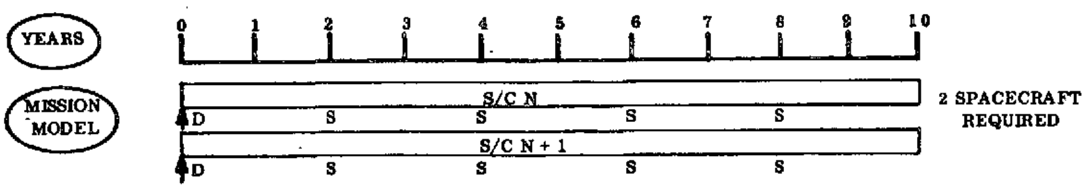


											TOTALS
SPACECRAFT COSTS	32.8	-	32.8	-	-	-	32.8	-	32.8	-	131.2
LAUNCH COSTS UP	3.62				3.31						
DWN	3.62				3.31						
TOTAL	7.24	-	7.24	-	6.62	-	7.24	-	7.24	-	35.6
FSS COSTS	1.3	-	.20	-	.19	-	.20	-	.20	-	2.1
SPMS COSTS	-	-	.5	-	.5	-	.5	-	.5	-	2.0
GROUND COSTS	.2									.2	2.0
REFURBISHMENT COSTS	6.6	-	6.6	-	13.2	-	6.6	-	6.6	-	39.6
								TOTAL COSTS			212.5

FIGURE 5-7
COST ANALYSIS - ON-ORBIT SERVICEABLE S/C - (6 YR ORBITAL LIFE)

The second option shown in Figure 5-8 requires only two spacecraft making this option more cost effective if it is realistic to maintain a spacecraft for ten years in orbit without a complete ground overhaul.

The program cost for the ten year spacecraft is 157 M dollars and is therefore the most cost effective option investigated for the nominal case.



LAUNCH SCHEDULE

											TOTALS
DELIVERY	2	-	-	-	-	-	-	-	-	-	2
SERVICE	-	-	2	-	2	-	2	-	2	-	8
RETRIEVE	-	-	-	-	-	-	-	-	-	-	-

COST ANALYSIS

SPACECRAFT COSTS	65.6	-	-	-	-	-	-	-	-	-	65.6
LAUNCH COSTS UP	3.22		3.31								
LAUNCH COSTS DWN	3.22		3.31								
LAUNCH COSTS TOTAL	6.44	-	6.62	-	6.62	-	6.62	-	6.62	-	32.9
FSS COSTS	1.3	-	.19	-	.20	-	.19	-	.19	-	2.1
SPMS COSTS	-	-	.5	-	.5	-	.5	-	.5	-	2.0
GROUND COSTS	0.2	←							→	0.2	2.0
REFURBISHMENT COSTS	-	-	13.2	-	13.2	-	13.2	-	13.2	-	52.8
								TOTAL COSTS			157.4

FIGURE 5-8

COST ANALYSIS - ON-ORBIT SERVICEABLE S/C - (10 YR ORBITAL LIFE)

A summary of the nominal case cost analysis is presented in Table 5-5 and indicates a clear advantage of the serviceable spacecraft modes over the expendable spacecraft mode. The choice of optimum serviceable spacecraft mode depends on the allowable on orbit spacecraft life and the impacts of the cost sensitivities discussed in later sections.

TABLE 5-5 NOMINAL CASE COST ANALYSIS SUMMARY

CASE	COST	
	M$	NORMALIZED
EXPENDABLE SPACECRAFT	319	(2.03)
GROUND SERVICED S/C (SINGLE LAUNCH) (DUAL LAUNCH)	188 210	(1.19) (1.33)
COMBINED ON-ORBIT & GROUND SERVICED S/C (1 SERVICE & RETURN) (2 SERVICES & RETURN)	192 189	(1.22) (1.20)
ON-ORBIT SERVICED SPACECRAFT (S/C LIFE 6 YRS & DISCARD) (S/C LIFE 10 YRS & DISCARD)	213 157	(1.36) (1.00)

- EXPENDABLE SPACECRAFT NOT COST EFFECTIVE.
- ON-ORBIT SERVICED S/C (WITH 10 YR LIFE) MOST COST EFFECTIVE FOR NOMINAL CASE.
- CHOICE BETWEEN SERVICE OPTIONS DEPENDS ON
  - ON-ORBIT LIFE OF SPACECRAFT
  - IMPACT OF COST SENSITIVITIES

5.3.2 COST SENSITIVITY ANALYSIS & IMPACTS

This section covers the cost impacts of the ranges of variables defined in table 5-4 of section 5.2.5. The summary of the cost impacts of the defined variables is presented in table 5-6 and discussed below.

5.3.2.1 High Refurbishment Cost Impacts

The major impact of increasing the refurbishment costs is the relative increase in the ground serviceable over the on-orbit serviceable options. In all cases the expendable spacecraft cost far exceeds the serviceable spacecraft costs, three of the four orbital serviceable spacecraft concepts show program costs less than the ground serviceable spacecraft concepts. The on-orbit serviceable spacecraft also improve in ranking over the combined ground and in-orbit serviceable concepts.

C - 2

TABLE 5-6
SHUTTLE MODE COST ANALYSIS COST SENSITIVITY SUMMARY

CASE	NOMINAL COST (SUCCESS CASE)	IMPACTS OF COST SENSITIVITIES									
		HIGH REF. COSTS	FULL LAUNCH COSTS	LOW S/C COSTS	HIGH S/C COSTS	1 FAILURE	2 FAILURES	3 FAILURES	Δ NON.REC COSTS (SERVICE)	HIGH GRD COSTS	PRORATED S/C COSTS
EXPEND SPACECRAFT	(2.03) 319	(1.74) 319	(2.25) 389	(1.86) 220	(2.11) 414	(1.92) 351	(2.01) 382	(2.10) 414	(1.90) 319	(1.82) 319	(2.03) 319
EXPEND SPACECRAFT (DUAL LAUNCH)	(2.03) 319	(1.74) 319	(1.97) 340	(1.86) 220	(2.11) 414	(1.92) 351	(2.01) 382	(2.10) 414	(1.90) 319	(1.82) 319	(2.03) 319
GROUND SERVICE SPACECRAFT	(1.19) 188	(1.26) 230	(1.49) 257	(1.16) 137	(1.22) 239	(1.21) 221	(1.22) 232	(1.23) 243	(1.14) 192	(1.18) 206	(1.06) 167
GROUND SERVICE SPACECRAFT (DUAL LAUNCH)	(1.33) 210	(1.34) 246	(1.32) 229	(1.29) 152	(1.36) 268	(1.33) 243	(1.33) 254	(1.35) 265	(1.27) 214	(1.30) 228	(1.07) 168
GROUND & ON ORBIT SERV. S/C (1 SERVICE & RETURN)	(1.22) 192	(1.20) 219	(1.18) 205	(1.20) 142	(1.24) 243	(1.19) 218	(1.18) 225	(1.18) 232	(1.21) 203	(1.20) 210	(1.08) 169
GROUND & ON ORBIT SERV. S/C (2 SERVICE & RETURN)	(1.20) 189	(1.16) 213	(1.22) 211	(1.18) 139	(1.21) 237	(1.17) 215	(1.17) 222	(1.16) 229	(1.19) 200	(1.18) 207	(1.06) 166
ORBIT SERVICE SPACECRAFT (6 YR LIFE)	(1.36) 213	(1.27) 232	(1.30) 225	(1.32) 156	(1.37) 269	(1.31) 239	(1.29) 246	(1.28) 253	(1.33) 224	(1.32) 231	(1.21) 191
ORBIT SERVICE SPACECRAFT (10 YR LIFE)	(1.00) 157	(1.00) 183	(1.00) 173	(1.00) 118	(1.00) 196	(1.00) 183	(1.00) 190	(1.00) 197	(1.00) 168	(1.00) 175	(1.00) 157

NORMALIZED COSTS SHOWN IN ( )

5.3.2.2 Full Launch Cost Impacts

When full shuttle launch costs are charged to the EOS program in place of sharing the shuttle charges with other programs, the advantage of dual launches for both the expendable and ground serviceable spacecraft become evident. The dual launch concept saves 49 M in the expendable spacecraft case and 28 M in the ground serviceable spacecraft case. The increased launch costs also improves the relative ranking of on-orbit serviceable spacecraft.

5.3.2.3 Spacecraft Recurring Cost Impacts

The choice of spacecraft recurring costs within the range of 20 to 40 M dollars has relatively little impact on the ranking of alternate servicing modes of operation. The lower spacecraft costs do however provide a slight improvement in ground servicing over on-orbit servicing.

5.3.2.4 Cost Impacts of Failures

The number of failures experienced during a ten year program has a significant impact on the selection of the optimum servicing mode. When failures were assumed the on-orbit serviceable spacecraft cost rose less than the costs of the ground serviceable spacecraft showing three of the four on-orbit serviceable concepts more cost effective than the ground serviceable concepts. A realistic comparison of the relative merits of the alternate servicing modes must consider this cost sensitivity factor.

The following assumptions were made to calculate the impacts of failures on each of the spacecraft cases considered.

1) Expendable Spacecraft

   A failure of an expendable spacecraft requires a new spacecraft, (28.8 M dollars) plus the costs to launch the new spacecraft

2) Ground Serviceable Spacecraft

   One additional spacecraft (at the cost of 30.3 M dollars) is required to allow continuous on-orbit operation of the system. It was assumed that a shuttle launch can be scheduled within 3 months and that the 3 month period with only one spacecraft operating in orbit is acceptable. The spare spacecraft is launched when a shuttle flight can be scheduled and the failed spacecraft returned to the ground. This failed spacecraft is refurbished (at a cost of 9.1 M dollars) and then operates as the new spare.

3) On-Orbit Serviceable Spacecraft

Spare modules of all on-orbit serviceable equipment (at a cost of 22.3 M dollars) are required in this mode of operation to ensure that on-orbit down time is maintained at a minimum. When a failure occurs a shuttle flight is scheduled to perform the replacement of the failed module and also perform any preventive maintenance. The failed module is returned via shuttle along with other modules replaced and refurbished (at a cost of 3 M dollars) for later use.

5.3.2.5 Non Recurring Costs of Servicing

Designing the spacecraft and shuttle support system to allow on-orbit servicing will require significant non recurring costs. These costs have been assumed as:

	Ground Serviceable Spacecraft	On-Orbit Serviceable Spacecraft	
Spacecraft Design & Integration	4 M	10 M	
Positioning Platform (FSS)	---	1 M	*(these costs are not chargeable to the EOS program)
Special Purpose Manipulator System	---	*	
Total	4 M	11 M	

When these non recurring costs are added to the on-orbit serviceable options their cost advantages over the ground serviceable spacecraft are reduced.

5.3.2.6 <u>High Ground Cost Impacts</u>

The cost spread between the expendable spacecraft and serviceable spacecraft options are so large that even an increase in the ground costs by a factor of ten for the serviceable spacecraft options show little impact on the relative costs.

5.3.2.7 Impact of Prorating Spacecraft Costs

In many of the nominal cases investigated the number of spacecraft required provide for spacecraft operation in excess of the assumed ten year program. In these cases, that option was penalized for the cost of the additional spacecraft without accounting for the additional lifetime of the system. If the spacecraft costs are prorated to account for the additional lifetime the costs of all options with the exception of the expendable spacecraft and the ten year lifetime on orbit serviceable spacecraft will decrease. This decrease is significant, although the ten year lifetime on-orbit serviceable spacecraft still remains the lowest cost option.

5.3.2.8 Summary

The most significant cost sensitivities investigated were:

- Refurbishment Costs
- Launch Costs
- Spacecraft Failures
- Non-Recurring Costs of Servicing
- Prorating Spacecraft Costs

These variables were re-investigated and combined sensitivities determined for the most realistic alternate values of the variables determined. (See Section 5.3.3.)

5.3.3 REVISED VARIABLES AND COST ANALYSIS

The re-investigation of the key variables identified previously show the following conclusions:

1) Refurbishment Costs

   A subsystem by subsystem investigation of the anticipated refurbishment costs indicate that the originally assumed refurbishment costs of 9.1 M dollars for ground refurbishment and 6.6 M dollars for in-orbit service are valid estimates of the refurbishment costs.

   Since this is a key cost area and previous studies for other applications have indicated higher refurbishment cost estimates this area will remain a variable in the updated cost analysis. The cost variation carried will remain as shown previously.

ground refurbishment (2 yr.)	9.1 M to 15.1 M
ground refurbishment (failure)	7.6 M to 12.6 M
on-orbit service (2 yr.)	6.6 M to 9.8 M
on-orbit service (failure)	3.3 M to 6.6 M

The lower cost figure is still considered as the most realistic estimate of anticipated refurbishment costs.

2) Launch Costs

The actual launch costs charged to EOS is most likely bounded by the costs determined using the shuttle trip charge formula supplied for this study and the full shuttle charge of 9.8 M dollars round trip. The nominal cost case will continue to use the shuttle trip charge formula while the alternate case will assume a cost averaged between the full trip charge of 9.8 M and the cost using the supplied formula.

3) Spacecraft Failures

It is imperative to include the impact of spacecraft failures on the shuttle mode utilization cost analysis. The impact of failures has been determined to be significant and it is unrealistic to assume no failures in a 10 year program with two spacecraft operating at all times. The revised nominal case will consider two spacecraft failures while the alternate case will assume three failures.

4) Non-Recurring Costs of Servicing

The non-recurring costs associated with the servicing modes must be included in the analysis if valid comparisons are expected between servicing and non-servicing modes. The non-recurring costs assessed against the alternate servicing modes are:

Ground Serviceable Spacecraft = 4.0 M

On-Orbit Serviceable Spacecraft = 11.0 M

These values will be used in both the revised nominal and alternate cost analysis.

The revised cost study has been separated into two independent analyses. The first analysis assumes the nominal cost impact of variables while the second analysis assumes the maximum realistic values of the variables. The costing assumption used in each analysis are summarized in Table 5-7. A summary of the two combined

TABLE 5-7
REVISED COSTING ASSUMPTIONS

	REVISED NOMINAL COSTS	ALTERNATE COST (MAX VARIABLES)
MISSION MODEL & ORBIT	SAME AS NOMINAL COST ASSUMPTIONS	
SPACECRAFT COSTS, WTS & LIFE GROUND SERVICE ON ORBIT SERVICE (2 YR) ON ORBIT SERVICE (FAILURE)	 9.1M 6.6M 3.3M	 15.1M 9.8M 6.6M
SHUTTLE COST & ACCOMMODATIONS	LAUNCH COSTS PER TRIP FORMULA (REVISED)	COST AVERAGED BETWEEN TRIP FORMULA & MAX COST
GROUND COSTS	SAME AS NOMINAL COST ASSUMPTIONS	
NUMBER OF SPACECRAFT FAILURES	TWO	THREE
ADDITION OF NON-RECURRING COSTS GROUND SERVICE ON-ORBIT SERVICE	 4.0M 11.0M	 4.0M 11.0M

REVISED NOMINAL CASE ALSO INVESTIGATED FOR PRORATED SPACECRAFT COSTS

sensitivity cost analyses is presented in Table 5-8. The results of the revised nominal cost analysis indicate that on-orbit servicing provides lower cost than ground servicing even when the spacecraft costs are prorated over a program length of greater than ten years. The expendable spacecraft mode of operation is nearly twice the cost of the lowest servicing option. When the maximum realistic values of variables are considered in the alternate cost analysis, the advantage of on-orbit servicing over ground servicing becomes more pronounced as seen by the last column of Table 5-8.

TABLE 5-8
REVISED COST SUMMARY (IMPACT OF COMBINED SENSITIVITIES)

CASE	NOMINAL COST M$				ALTERNATE COST M$ (MAX VARIABLES)	
	OPTION #1		OPTION #2			
	•NOM. REFURB. •2 FAILURES •TOTAL S/C COSTS •N.R. SERVICE COSTS	NORMALIZED COST	•NOM REFURB. •2 FAILURES •PRORATED S/C COSTS •N.R. SERVICE COSTS	NORMALIZED COST	•HIGH REFURB. •3 FAILURES •TOTAL S/C COSTS •MOD. HIGH LAUNCH COSTS •N.R. SERVICE COSTS	NORMALIZED COST
EXPENDABLE SPACECRAFT (SINGLE LAUNCH)	382	1.90	382	1.90	459	1.71
EXPENDABLE SPACECRAFT (DUAL LAUNCH)	382	1.90	382	1.90	428	1.59
GROUND SERVICE SPACECRAFT (SINGLE LAUNCH)	236	1.17	212	1.05	381	1.42
GROUND SERVICE SPACECRAFT (DUAL LAUNCH)	258	1.28	213	1.06	352	1.31
GROUND & ON-ORBIT SERV S/C (1 SERVICE & RETURN)	236	1.17	215	1.07	306	1.14
GROUND & ON-ORBIT SERV S/C (2 SERVICE & RETURN)	233	1.16	213	1.06	300	1.12
ORBIT SERVICE SPACECRAFT (6 YR LIFE)	257	1.28	237	1.18	317	1.18
ORBIT SERVICE SPACECRAFT (10 YR LIFE)	201	1.00	201	1.00	269	1.00

5.3.4 COST ANALYSIS SUMMARY

The following conclusions can be made from the preceding cost analysis:

- The expendable spacecraft is the least cost effective of all the cases considered.
- The on-orbit serviced spacecraft (assumed 10 year life on-orbit) is the lowest cost for all cases considered.
- The cost differential between the combined on-orbit and ground serviced spacecraft is negligible for the nominal cost options.
- The on-orbit serviced spacecraft are most cost effective when spacecraft failures are considered and refurbishment costs increase as indicated in the (max. variable) cases.

APPENDIX A
SHUTTLE ORBIT TRADES

1.0 INTRODUCTION AND SUMMARY

The selection of the recommended shuttle delivery, retrieve or service orbit involves cost and performance trades for expendable launch vehicles, on-board propulsion systems and the Space Shuttle. The trades also consider recovery at mission altitude or a lower altitude and evaluate the relative advantages of elliptical or circular shuttle recovery orbits. The recommended shuttle orbits for a mission orbit of 418 nm are as follows in Table A-1. The shuttle retrieve altitude for Delta and Titan launched spacecraft was selected to be 330 nm. This altitude represents a compromise between minimizing shuttle charges (lower altitudes preferred) and minimizing the weight impact on the spacecraft on board propulsion system to make large Δ V orbit transfer burns. The selected altitude occurs just below the altitude where a second OMS kit must be added to shuttle and thus is an optimum point for allowable payload weight.

TABLE A-1
RECOMMENDED SHUTTLE ORBITS

Launch Vehicle	Recommended Shuttle Orbit For		
	Delivery	Retrieve	Service
Delta	---	330 nm	---
Titan	---	330 nm	TBD
Shuttle	250 nm	250 nm	250 nm

The recommended shuttle orbit for a Titan service mission cannot be determined unless the detail weight of the spacecraft is known. It is most likely that a Titan launched servicing mission will be weight critical and therefore the recommended shuttle service altitude will be 330 nm.

2.0 SHUTTLE SERVICE ORBIT IMPACTS

The relative advantages and impacts of circular and elliptical servicing orbits have been investigated in detail by Jerome Bell of the JSC Mission Analysis Branch. Four documents have been issued to summarize his findings. These documents which were supplied as reference data at the beginning of the EOS study are:

JSC-08596 "Placement of the Goddard Earth Observation Satellite into its Operational Orbit after Orbital Servicing" January 28, 1974.

JSC-08599 "Effects of an Elliptical Servicing Orbit on Orbiter Rendezvous with the Goddard Earth Observation Satellite" January 29, 1974.

JSC-08686 "EOS Maneuvering to a Shuttle Compatible Servicing Orbit prior to Shuttle Lift-Off" February 4, 1974.

JSC-08878 "Preliminary Representation Mission Profile and Performance Analysis for a Typical EOS Servicing Mission" March 7, 1974.

The two alternate EOS Servicing orbits investigated in the previously referenced reports are:

1) a 307 nmi circular phase repeating orbit
2) a 490 by 124.5 nmi elliptic phase repeating orbit

The circular phase repeating orbit is preferred over the elliptical orbit for the following reasons:

1) Elliptical servicing will require added crew training and more detail analysis for the more complex rendezvous case
2) The elliptical orbit imposes geographic constraints on the time of the EOS deboost maneuver allowing less flexibility in achieving the required phasing relationships
3) The elliptical orbit option limits the flexibility of accommodating variations in shuttle performance since perigee can not be lowered below the presently assumed 124.5 nm

(Therefore, elliptical shuttle orbits will only be used as a backup in case a failure of the spacecraft propulsion system does not allow the spacecraft to return to the recommended shuttle circular orbit)

Shuttle performance for circular sun synchronous orbits is presented in Figure A-1. The top two lines on this figure represent the shuttle no-rendezvous and rendezvous performance and were obtained from JSC 07700 volume XIV Revision B "Space Shuttle System Payload Accommodations". These values will be used in establishing a parametric analysis to determine the optimum shuttle service altitude. The lower two lines on the figure establish net payload capabilities by subtracting Flight Support System (FSS) and Special Purpose Manipulation System (SPMS) weights.

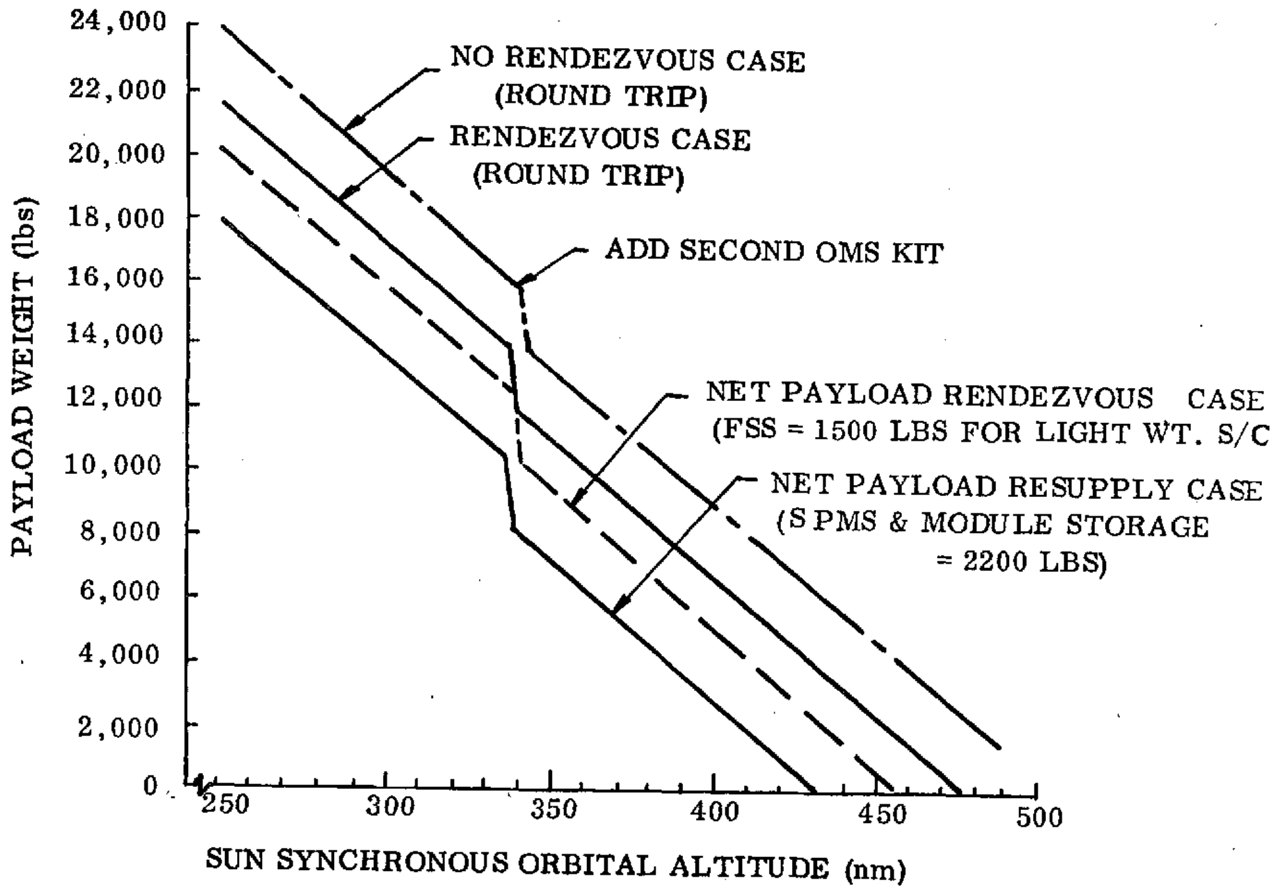


FIGURE A-1
SPACE SHUTTLE PERFORMANCE

2.1 EFFECT OF SHUTTLE SERVICING ON MISSION ORBIT

In order to service an EOS spacecraft in orbit, the satellite must be lowered from its operational altitude to one in the neighborhood of 300 nm. Launch of the shuttle into this lower altitude and use of an on-board EOS hydrazine propulsion system for maneuvering to/from the operational orbit is shown in sections 3.0 and 4.0 of this appendix to be the most cost-effective method of launching/retrieving/servicing EOS satellites. Because the satellite will be in an orbit other than its mission orbit during the servicing period, its mission ground track will be altered reflecting the differences in period and nodal regression between the two orbits. Two effects result:

1) a shift in the ground trace - the orbit is still repeatable following servicing but does not repeat in the same place as it did prior to servicing
2) a shift in the node - the orbit is still sun synchronous but the Beta angle has been shifted.

Just how significant the difference will be between the pre- and post-service orbits depends on the following:

- the duration of time spent at the lower altitude - the longer the time the greater the effect on the mission orbit
- the degree of optimization utilized in planning and executing the orbit transfer maneuvers
- the amount of propulsion system capability available to re-establish the initial conditions.

Analysis performed at JSC* has shown that there is an optimum point occurring periodically (roughly once a day) from which the initial mission ground trace can be re-established with minimum propellant usage. The option exists to have either the shuttle or the EOS perform the correction maneuvers. In either case sufficient propellant will be available for the maneuvers such that the preservicing mission ground trace can always be re-established.

*Placement of the Goddard Earth Observatory Satellite into its Operational Orbit after Servicing, JSC Internal Note No. 74-FM-4. January 28, 1974.

No similar periodic optimum point occurs for the nodal error however; the longer the satellite is at the lower orbit, the greater the shift in the node, hence the greater the propulsion capability required to make the correction. For nominal servicing periods of four** days the shift in the node and resultant change in Beta angle are small, typically 0.5 degree for the Beta angle, and generally would not require compensation. If compensation is desired, node biasing techniques can be utilized and the propellant requirements are well within the capability of the EOS hydrazine propulsion system. For cases where the servicing periods become extensive (2-3 weeks in a contingency case for example) a point will be reached where the propulsion system will not be capable of directly compensating for the change in node. For this later case, the Beta angle will have changed by some amount which is a function of the actual time spent at the lower altitude. The change could amount to several degrees. This may not be significant to many payloads, but assuming it is*, a long-term corrective solution exists to re-establish the initial node (and hence Beta angle). This solution utilizes a comparatively small amount of propellant, and involves placing the spacecraft in a slightly non-sun synchronous orbit upon return to mission altitude. This orbit will cause a slow drift of the node back toward its desired position, ideally to be back at the desired position at the time of the next servicing period (nominally two years). Even though the inclination and altitude of the non-sun synchronous orbit will be slightly different than prior to servicing, the resultant ground trace will nevertheless be identical to the pre-servicing one.

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* Consider the case of a 10 year minimum life with service every two years. An uncorrected node error will accumulate over the service periods and, assuming several long-service periods, could grow to a value which would impact payloads and the power subsystem.

** Preliminary Representative Mission Profile and Performance Analysis for a Typical Earth Observatory Satellite Servicing Mission, JSC Internal Note No. 74-FM-17, March 7, 1974.

2.2 MULTIPLE SATELLITE SERVICING

There are two separate cases to be considered for multiple satellite servicing:

1. the spacecraft are in the same mission orbit, e.g., two EOS satellites at 418 nm altitude phased one-half orbit apart.
2. the spacecraft are at different altitudes but with inclinations "close enough" to be serviced by a single shuttle flight.

In the first case either one or both of the satellites can be serviced by one shuttle orbiter. If only one satellite is serviced, it must be returned to its pre-service mission orbit as described in the previous section since the relative phasing between the spacecraft is key to the ground coverage interval.

If both spacecraft are to be serviced they both would likely be maneuvered to a lower servicing orbit prior to shuttle launch. Since the satellites have different node times, optimization of this maneuver would be required using node biasing techniques to insure their return to the same relative phasing following service. Service of the first spacecraft would proceed in the same fashion as if only one spacecraft were involved with a typical time in lower orbit of four days. It could then be returned to operational altitude. Shuttle maneuvering to the second spacecraft and subsequent service would then begin. Further tradeoff studies are required to determine if the second spacecraft should be maneuvered near the first to minimize shuttle maneuvers or if all chase and rendezvous maneuvers should be performed by the shuttle. In either case rendezvous can be achieved; one approach may be better in terms of propellant usage (spacecraft, shuttle or both) and time. Again in either case, a nominal servicing of the second spacecraft could be completed within two days of servicing of the first. The second spacecraft would then be returned to its operational orbit, phased properly with the first spacecraft. No significant problems are envisioned in the servicing of two spacecraft which have the same mission orbit differing only in node time.

Success in the second case, where two spacecraft are at different altitudes and inclinations, is strictly determined by the capability of the propulsion systems, the shuttle, the spacecraft or both, to supply the cross plane change capability to align the inclinations at the servicing altitude and return to the original inclinations (and altitudes) following service. The ability to align the inclinations is a function of many variables, the altitudes, inclinations, weights, and propulsion system capabilities of the satellites, the servicing orbit altitude plus the propulsion system capability available in the shuttle for that particular mission.

Given that the orbital planes can be aligned, servicing would proceed in a similar fashion to case 1. Return of the satellites to their pre-servicing mission ground traces may or may not be a requirement depending on the missions involved. It can be assumed that phasing of the two satellites will not be required since satellites at different mission altitudes are generally not mission related.

3.0 PRE-SHUTTLE ERA LAUNCHED SPACECRAFT

The baseline pre-shuttle launch vehicle for EOS is the Delta 2910. The allowable spacecraft weight and volume is severely constrained for an EOS Delta 2910 launch and therefore shuttle retrieval and ground resupply is the only viable resupply option. On-orbit resupply with its associated weight penalty of 400 to 500 lbs. would place the spacecraft weight well in excess of the Delta 2910 launch capability. Thus, the major unknown for pre-shuttle launched spacecraft is the choice of the optimum shuttle retrieval altitude. The major variables involved in this choice are the shuttle retrieval costs and the allowable spacecraft weight (constrained by the Delta launch). If the shuttle retrieval altitude is selected as the mission altitude an orbit transfer capability is not required on the spacecraft. If an orbit transfer capability is provided on the basic spacecraft (large Δ V engines and increased propellant capability) the spacecraft can transfer to an altitude lower than the mission altitude for rendezvous with the shuttle at an altitude where shuttle has increased payload capability. The shuttle cost to EOS can then be reduced by sharing the shuttle charges with other payloads that can be delivered simultaneously by the shuttle. A formula that includes payload sharing of shuttle charges has been supplied for the EOS Study. Using this formula and other assumptions contained in Table A-2, a parametric analysis of transportation costs as a function of mission altitude and shuttle retrieval altitude can be performed with the results summarized in Figure A-2. The upper curve for Delta delivery and shuttle

TABLE A-2
SHUTTLE CHARGES AND COSTING ASSUMPTIONS
(DELTA LAUNCH, SHUTTLE RETRIEVE)

SHUTTLE CHARGES

Payload up and down cost = 9.8M max. (4.9M up and 4.9M down)

Payload up cost = 4.9M (load factor)

$$\text{where load factor} = \frac{\text{Payload up Weight}}{(.78)\ \text{Shuttle Payload Capability}}$$

Payload down cost = 4.9M (load factor)

$$\text{where load factor} = \frac{\text{Payload down Weight}}{(.78)\ \text{Shuttle Payload Capability}}$$

Cargo Manifest - Share Payloads

- Materials Processing Module
- Life Sciences Module
- Short Pallet
- Hitch Hiker Pallet

Note: The shuttle charge formula has been modified by GE to account for shuttle loading inefficiencies by adding a factor of .78 to the formula

COSTING ASSUMPTIONS

Delta costs - $6M

Propulsion System Costs

RCS & OA = $.5M
RCS, OA & OT = $.6M

Δ Cost for orbit transfer = $.1M

EOSA wt = 2200# (minus propulsion system)

Shuttle support wt = 1500#

Δ Cost for added reliability = $.25M (orbit transfer case) retrieval at mission altitude shows that the transportation cost is a function of mission altitude and that the mission altitude is limited to approximately 450 nm due to the shuttle retrieval performance as defined in Figure A-1. The lower curves for alternate shuttle retrieval altitudes defined at the right hand portion of Figure A-2 indicate the cost savings that can be achieved by adding an orbit transfer system to the basic spacecraft propulsion system. This addition also makes the transportation cost relatively insensitive to the mission altitude. It should be noted that significant cost savings are achieved in reducing the shuttle retrieval altitude from 390 to 360 and from 360 to 330 nm, but very little additional cost is saved in reducing the shuttle retrieval altitude further. Since the weight allowable on the spacecraft is limited and therefore the weight for orbit transfer fuel is also limited the shuttle retrieval altitude of 330 nm is selected as preferred for a Delta launched spacecraft giving a total vehicle weight of 2420 lbs. (including propulsion) for a mission orbit of 418 nm. As shown on the figure a cost savings of 2.5 M$ is achieved by lowering the shuttle retrieve altitude from the mission altitude of 418 nm to 330 nm. A summary of the cost savings of

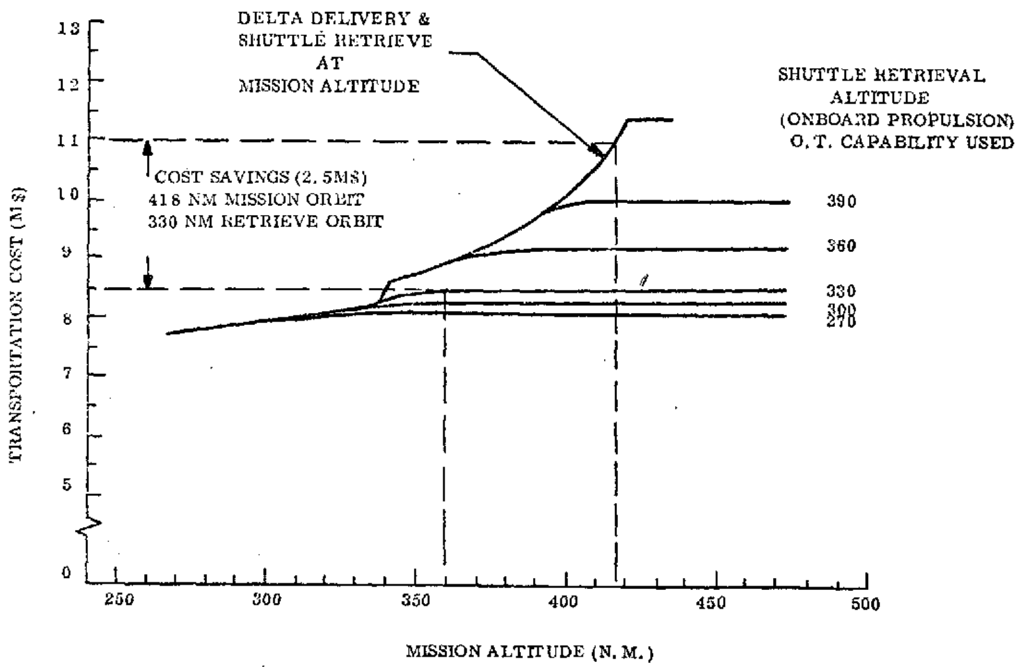

FIGURE A-2
DELTA LAUNCHED SPACECRAFT TRANSPORTATION COST

retrieval at 330 nm over retrieving at mission altitudes of 420, 400, 380 and 360 is presented in Table A-3. An important point to remember is that the capability always exists to retrieve the spacecraft at the mission altitude if a failure in the propulsion system precludes returning the spacecraft to the desired lower orbit. This data for Delta 2910 is typical for an expendable launch vehicle with a limited payload capability and the results would be similar for Delta 3910 and Titan IIIB.

TABLE A-3
TRANSPORTATION COST SAVINGS FOR SHUTTLE RETRIEVE AT 330 n.m.
(DELTA LAUNCH)

	Transportation Cost M$		
Mission Orbit	Shuttle Retrieve @ Mission Orbit	Shuttle Retrieve @ 330 n.m.	Transportation Cost Savings for Retrieve @ 330 nm
420	11.3	8.5	2.8
400	10.2	8.5	1.7
380	9.4	8.5	0.9
360	9.0	8.5	0.5

4.0 SHUTTLE LAUNCHED SPACECRAFT

When shuttle becomes operational for launches to sun synchronous orbits the EOS spacecraft will no longer be severely constrained in its allowable launch weight as it is for a Delta launch. Lifting this weight restriction allows a re-examination of the preferred delivery, retrieval or servicing orbit for EOS. The shuttle charges and costing assumptions for the full shuttle era case are summarized in Table A-4. The shuttle charge formula used is identical to the formula used in the Delta launched case and defined in Table A-2. The EOS weight for a shuttle launch has been assumed for this analysis to be 4000 lbs. while the modules required for resupply were assumed to weigh 2500 lbs. Three tradeoff curves were generated for the shuttle launch case.

TABLE A-4
SHUTTLE CHARGES AND COSTING ASSUMPTIONS
(SHUTTLE LAUNCH)

Shuttle Charges - See Table A-2

Costing Assumptions

EOS weight = 4000# (minus propulsion)

Servicing mission wt = 2500# (minus propulsion)

Shuttle support wt = 2000# (delivery or retrieval)
= 3200# (servicing mission)

Propulsion System Costs

RCS & O.A. = .5 M$
RCS, O.A. & O.T. = .7 M$ } Δ Cost for orbit transfer = .2 M$

Cost for added reliability = .25 M$ (orbit transfer case)

The first tradeoff curves, shown in Figure A-3 present the cost trades for a shuttle launch and retrieve at an altitude below mission altitude. It is obvious from these curves that there is a considerable cost savings when shuttle delivers the spacecraft to a low altitude and also retrieves the spacecraft at the low altitude. Cost savings of between 5.7 M$ and 3.5 M$ are shown for mission orbits between 400 and 360 nm and a shuttle orbit of 250 nm. The shuttle orbit of 250 nm was selected as a realistic altitude that provides meaningful transportation cost savings while not placing excessive orbit transfer requirements on the basic spacecraft propulsion system.

The second tradeoff curves, shown in Figure A-4 assume a shuttle delivery to a lower than mission orbit altitude but consider a shuttle retrieval at the mission altitude. This case shows the cost savings if a spacecraft failure would preclude firing the on-board propulsion system to lower the spacecraft altitude for retrieval by shuttle. Significant cost savings in the range of 1.7 to 2.8 M$ are still shown for this case.

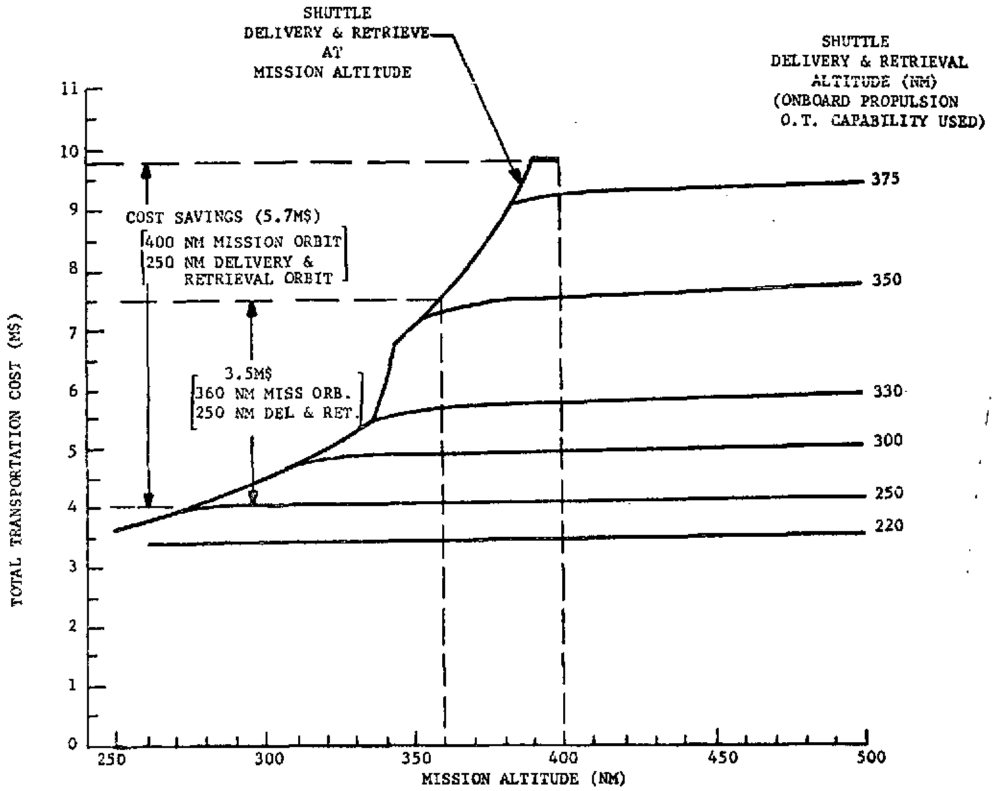

FIGURE A-3
SHUTTLE LAUNCHED SPACECRAFT TRANSPORTATION COST

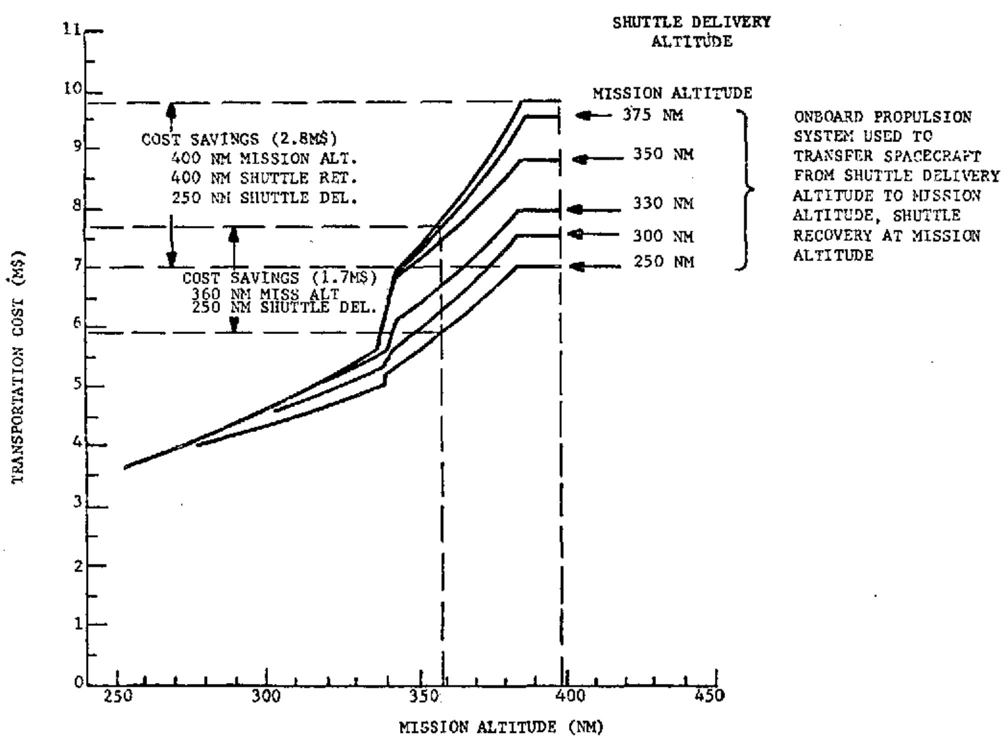

FIGURE A-4
SHUTTLE LAUNCHED SPACECRAFT TRANSPORTATION COST
(LAUNCH TO LOW ORBIT, RECOVERY @ MISSION ORBIT)

The final tradeoff curves which are shown in Figure A-5 illustrate the cost tradeoffs for a servicing mission and again indicate significant transportation cost savings when the servicing mission is performed at 250 nm instead of the mission altitude.

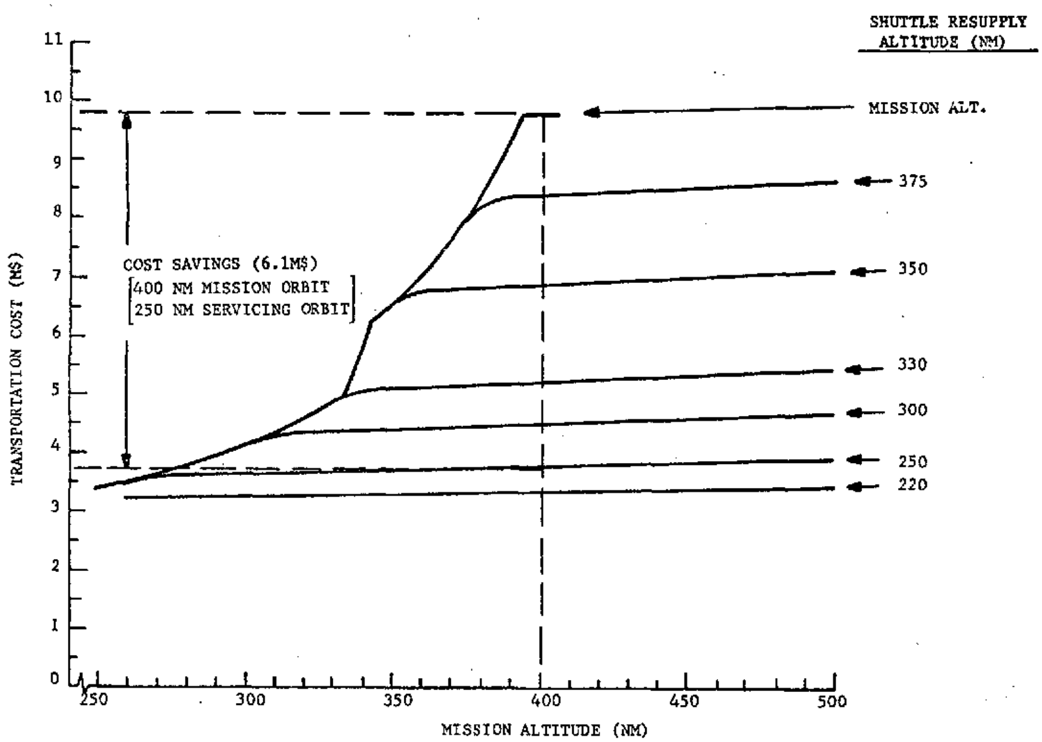

FIGURE A-5
SHUTTLE LAUNCHED SPACECRAFT TRANSPORTATION COST
(SERVICING MISSION)

The shuttle delivery, retrieval and service altitude of 250 nm is recommended for a shuttle launched spacecraft and involves a compromise between shuttle trip charges and on-board propulsion system weight and complexity. The cost savings involved in the two shuttle launch and retrieve cases is summarized in Table A-5.

TABLE A-5
TRANSPORTATION COST SAVINGS FOR SHUTTLE LAUNCH & RETRIEVE

Mission Orbit	Transportation Costs (M$)			Transp. Cost Savings (M$) using on-board prop.	
	Shuttle delivery & retrieve @ Mission orbit	Shuttle delivery & retrieve @ 250 nm	Shuttle delivery @ 250 & retrieve at Mission alt.	Delivery & retrieve @ 250	Delivery @ 250 retrieve @ Mission alt
400	9.8	4.1	7.0	5.7	2.8
380	9.0	4.1	6.8	4.9	2.2
360	7.6	4.1	5.9	3.5	1.7

APPENDIX B

SAFETY CONSIDERATIONS

1.0 INTRODUCTION

This section discusses the compatibility of the EOS design with the preliminary Shuttle Safety requirements. Although all current automated spacecraft programs are concerned with safety during ground flow operations, the use of the Shuttle as a launch/delivery system requires that this concern be extended into the flight phases of the mission. Potentially, this added set of requirements could produce an impact on all aspects of the system design and development. Preliminary analysis, however, indicates a modest cost and weight penalty based on the current state of Safety requirements definition. Clearly, the implementation of flight safety analysis and design provisions must be included in the first flight article program. The results of a preliminary safety review indicate that only relatively minor design modifications will be required to "safe" a Delta-launched EOS for a Shuttle launch. Potential impacts include increased wall thickness for propellant tanks, battery cell or case design and the addition of the caution and warning system.

2.0 SHUTTLE SAFETY REQUIREMENTS

The responsibility for issuing Shuttle requirements and guidelines for safety was recently transferred from the JSC Shuttle Project Office to the Office of Reliability, Quality Assurance, and Safety at NASA Headquarters. As a result, the data in Volume XIV, JSC 07700, Shuttle Payload Accommodations, for this area is virtually non-existent. The issuance of a requirements and guidelines document from NASA Headquarters has not been made general yet, and is now undergoing internal review. In its absence, reliance has been placed on Apollo and Skylab program practices and informal discussions with safety personnel at JSC and Headquarters.

In general, the two overall Shuttle safety objectives may be inferred as:

(a) Payload suppliers shall eliminate from their designs any conditions which may cause injury to a crewman or interfere with Shuttle subsystem operations; and

(b) Payload suppliers shall identify any residual hazardous condition inherent in their system and the preventive measures taken to contain the effects of such conditions.

It is important to note that the inherent design of virtually all payloads will contain elements which could conceivably be hazardous to the Shuttle and/or Shuttle crew. For example, nearly every automated spacecraft will contain a pressurized vessel. By definition, a pressure vessel poses a potential hazard whose probability of failure can be reduced, but not eliminated. The expected Shuttle safety guidelines recognize this problem and therefore permit the control and containment of potential hazards as a substitute in those cases where it is not feasible or cost effective to totally eliminate them.

3.0 POTENTIAL EOS HAZARDS ANALYSIS

A preliminary analysis was conducted of the EOS to identify potential sources of hazardous conditions. Table B-1 contains a list of those found and a summary statement of the disposition of each. The potential hazards listed in this table are limited to those which present a reasonable, i.e., single failure path problem. It can be seen from the Disposition column that all of these have been treated to some extent in the preliminary design phase. Those which still present some residual danger have been identified for continuous monitoring via the caution and warning system.

Premature deployment of the solar array, TDRSS antenna, or the EOS itself may present a serious problem, particularly during the Ascent phase. In general, however, we feel these are low probability events which are well controlled by the appropriate design practice. The inability to stow the array or antenna is a more likely contingency since there are more potential failures to cause it, and because these actions will be attempted after lengthy exposure to the space environment. For these and some other mechanical failures we have suggested the use of redundancy, larger design margins and thorough testing as potential solutions. EVA is considered as a back-up. Its utility has been shown clearly in Apollo and Skylab missions and should be considered in contingency situations.

TABLE B-1
SUMMARY EOS HAZARDS ANALYSIS

SUBSYSTEM	POTENTIAL HAZARD	LIKELIHOOD OF OCCURENCE	FAILURE EFFECTS	DISPOSITION	IMPACT
STRUCTURE/ MECHANICAL	1. EOS/RETENTION CRADLE ATTACHMENT BREAK	PROBABLY LIMITED TO RARE CASE OF SHUTTLE "CRASH" LANDING	PROBABLY CATASTROPHIC WITH SERIOUS INJURY TO OR LOSS OF CREW	DESIGN TO 9G (+X) AND 4.5G (+Z)	NO REAL IMPACT - STANDARD DESIGN FROM AIRCRAFT APPLICATIONS
	2. EXTERNAL COMPONENT BREAKS LOOSE (I.E., ARRAY OR ANTENNA)	RELATIVELY LOW	ALL COMPONENTS ARE LOW MASS - THUS PROBABLY NO CATASTROPHIC EFFECTS (NO PENETRATION OF FORWARD BULKHEAD)	DESIGN TO RESIST SHEAR LOADS	NO IMPACT - DESIGN STANDARD
	3. SOLAR ARRAY OR TDRS ANTENNA BOOM FAILS TO RETURN TO STOWED POSITION	MODERATE (CHAIN OF LOW FAILURE COMPONENTS)	(A) CANNOT RETRIEVE SPACECRAFT (ORBITER CARGO BAY DOORS CANNOT BE CLOSED) (B) INTERFERES WITH ON-ORBIT RESUPPLY	REDUNDANT DESIGN OF CRITICAL COMPONENTS - ALTERNATES: DESIGN FOR APPENDAGE JETTISON, DISCARD S/C, EVA ASSISTANCE	MODERATE COST LOW WEIGHT TO INCORPORATED REDUNDANCY; HIGH COST TO DISCARD S/C IN EVENT OF FAILURE
	4. MODULE FAILS TO RELEASE OR MODULE FAILS TO REINSERT	RELATIVELY LOW	CANNOT REPAIR FAILED MODULE OR PERFORM ON-ORBIT RESUPPLY - POTENTIAL LOSS OF VEHICLE IF IT CANNOT BE RETURNED TO GROUND, OR POTENTIAL LOSS OF SINGLE MODULE	(A) TECHNOLOGY RESEARCH PROGRAM (B) GOOD DESIGN PRACTICES (C) POTENTIAL EVA BACK-UP FOR MODULE LATCH FAILURE	MODERATE COST FOR GROUND TEST PROGRAM TO VERIFY DESIGN; POTENTIAL SIGNIFICANT COST IF FAILURE OCCURS
	5. PREMATURE DEPLOYMENT OF SOLAR ARRAY OR TDRS ANTENNA BOOM	RELATIVELY LOW	COULD PREVENT EOS DEPLOYMENT - AT WORST COULD CAUSE DAMAGE TO ORBITER CARGO BAY INTERIOR	(A) REMOTE SAFING FOR ALL PYRO ACTIVATION (B) CAUTION AND WARNING CANDIDATE	LOW COST AND WEIGHT DELTAS
	6. PREMATURE DEPLOYMENT OF EOS (FAILURE OF FSS)	RELATIVELY LOW	COULD PREVENT EOS SEPARATION FROM CARGO BAY - COULD CAUSE SEVERE DAMAGE TO CARGO BAY INTERIOR	(A) REDUNDANT DESIGN (B) CAUTION AND WARNING CANDIDATE (C) MISSION ABORT	LOW COST AND WEIGHT DELTAS
ELECTRICAL	1. PREMATURE RELEASE OF IN-FLIGHT UMBILICAL OR FAILURE TO REMATE FOR SERVICE/RESUPPLY	VERY LOW	LOSS OF POWER OR LOSS OF DATA/COMMAND LINK TO ORBITER - LOSS OF CAUTION AND WARNING MONITORING	(A) LARGE DESIGN MARGIN/COMPLETE GROUND TEST (B) REDUNDANT CONNECTORS (C) MISSION ABORT	LOW COST AND WEIGHT DELTAS
	2. BATTERY CASE BURST	VERY LOW	COULD PRODUCE DAMAGE IN CARGO BAY FROM BURST CASE - ESPECIALLY SERIOUS WHEN EOS IS ELEVATED OUT OF CRADLE	(A) BATTERY TEMPERATURE CANDIDATE FOR CAUTION AND WARNING (B) DESIGN STRONGER CASE OR BURST VALVE PRESSURE RELIEF	MODERATE COST IMPACT IF DESIGN CHANGES ARE NECESSARY
	3. SHORT CIRCUIT, OVER VOLTAGE, HIGH CURRENT, AND OTHER ELECTRICAL ANOMALIES	MODERATE TO LOW	NO DIRECT EFFECTS ON ORBITER, BUT FAILURE DURING DEPLOYMENT COULD PROPOGATE - FAILURE AT ANY TIME PRIOR TO SEPARATION EFFECTS ORBITER OPERATIONS (NOTE PROBABLE LOSS OF CAUTION AND WARNING MONITORING)	(A) MISSION ABORT (B) REDUNDANT CIRCUITS - FUSE PROTECTION (C) CAUTION AND WARNING CANDIDATE	LOW WEIGHT AND COST DELTAS - MOST PROVISIONS ALREADY INCLUDED
C&DH	1. FAILURE IN COMMAND DECODER (PRIMARY OR BACK-UP)	LOW	LOSS OF PRIMARY CAPABILITY TO COMMAND EOS IN RESPONSE TO OFF-NOMINAL CONDITIONS - SECOND FAILURE COULD CAUSE CATASTROPHIC CONDITION	(A) FAIL SAFE DESIGN PRACTICE (B) ADD BACK-UP COMMAND DECODER (C) MISSION ABORT RISK (D) CAUTION AND WARNING MONITORED	MINOR IMPACTS - GOOD DESIGN PRACTICE
PROPULSION	1. LEAK IN HYDRAZINE SYSTEM	LOW TO MODERATE	DAMAGE TO CARGO BAY INTERIOR - POTENTIALLY CATASTROPHIC EFFECTS MAY OCCUR IF SUFFICIENT HYDRAZINE IS LEAKED AND NOT EVACUATED	(A) HYDRAZINE ONLY PRESENT IN TANKS PRIOR TO SEPARATION (B) CAUTION AND WARNING MONITORED (C) MISSION ABORT POSSIBLE	SOME EXTRA COST AND WEIGHT DELTAS TO INSURE APPROPRIATE SAFETY LEVELS FOR SHUTTLE
	2. HYDRAZINE TANK RUPTURE	VERY UNLIKELY	PROBABLY CATASTROPHIC EFFECTS WITH LIKELY INJURY TO CREW AND SERIOUS DAMAGE TO ORBITER	(A) LOW PRESSURE SYSTEM (BLOWDOWN) (B) CAUTION AND WARNING MONITORED	GOOD DESIGN PRACTICE - VERY SMALL WEIGHT/COST DELTA FOR CAUTION AND WARNING
	3. INADVERTANT THRUSTER FIRING IN/NEAR CARGO BAY (ONLY RELEVANT FOR RETRIEVAL MISSIONS)	LOW TO MODERATE	TOTALLY DEPENDENT ON DURATION OF THRUSTER "ON" STATE AND SPECIFIC LOCATION/ATTITUDE OF VEHICLE - POTENTIALLY CATASTROPHIC IF DAMAGE OCCURS TO CARGO BAY DOORS OR ANY OTHER COMPONENT WHICH INTERFERES WITH DOOR CLOSING	(A) FAIL OFF DESIGN OF THRUSTER VALVES (B) PRE-DOCKING SAFETY CHECK (C) PROPELLANT DUMP PRIOR TO SAMS CAPTURE	MOSTLY PROCEDURAL - LOW COST/WEIGHT DELTAS

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There has been some concern expressed relative to the safety of a hydrazine propulsion/pneumatics system. From discussions we have had with NASA personnel, it has been concluded that such a system can be designed to meet shuttle safety requirements. An integral, all-hydrazine system operating in a blow-down mode appears to offer some safety advantages over pressure regulated or hybrid system designs. The major design feature which provides this favorable comparison is the low pressure nature of the system. The use of the blowdown mode of operations eliminates the need for a high pressure tank in the system and substantially reduces the likelihood of a tank rupture. The current state-of-the-art in propellant tank and plumbing seals also makes the probability of a leak very unlikely. The baselined hybrid system must also consider the safety aspects of the solid rocket engines and their ignition systems.

An additional safety factor in the Propulsion subsystem is provided by the operational use of the system. The approach is to "pressurize" the system, i.e., release hydrazine into the manifold downstream of the propellant tank, after the EOS has been separated from the Orbiter. This approach eliminates the potentially catastrophic effects of an "on" thruster command during EOS delivery. The problem can also exist following Orbiter recapture of the EOS for return to the ground or on-orbit servicing. However, the operational approach of fully depleting the tanks (or depleting the lines downstream of a shut-off valve) just before recapture, effectively eliminates this problem.

APPENDIX C

CONTAMINATION AND THERMAL CONTROL

1.0 INTRODUCTION AND SUMMARY

A review has been conducted of the compatibility of the EOS with the Shuttle induced environment. Two major areas of concern emerged from this review: sensor contamination and thermal control. These two areas are reviewed below. Other aspects of the environment, including vibration, acoustics, acceleration, and shock (discussed in section 2.2 of the main body of this report) were not considered problem areas. The EMI/EMC environment compatibility was not reviewed since Shuttle data is not currently available.

Many of the instruments which may be employed in the EOS program are sensitive to contamination. The control of gaseous and particulate contaminants is considered a problem with the Shuttle program at this time due to two primary factors. First, the definition of specific procedures and facilities is in its earliest stages and there are many unknowns. What preliminary data does exist, however, seems to suggest that contamination will be difficult to control. The second factor which suggests concern is the fact that multi-payload mission sharing will be a fact of life on the Shuttle where it is only a rarity at present. The sharing of a flight substantially increases the potential opportunities for foreign materials to be introduced into the cargo bay, and also increases the sources of contamination after initial payload closeout at the Orbiter Processing Facility (OPF). The nature of these contaminants, their effect on the EOS instruments, and potential approaches for their control or containment are discussed in Section 2.0.

Section 3.0 deals with the Shuttle cargo bay thermal environment. Two potential problems exist, one during the prelaunch phase and the other during the entry/post landing phase. It is concluded that the EOS can be thermally integrated with the Shuttle by selecting spacecraft orientations in the cargo bay to limit environmental effects on temperature critical components and limiting the maximum ground conditioning air temperature. Components such as batteries should be located away from the upper aft section of the payload bay during reentry. This criteria is met by the baseline design since the electrical power module is located in the lower portion of the payload bay.

2.0 SHUTTLE CONTAMINATION

This section describes the Shuttle environment as it pertains to contamination, the potential problems this environment creates for the EOS, and some initial concepts for solutions. Reference is made below to various cleanliness classes: 100, 10,000, and 100,000. These levels are defined in Figure C-1.

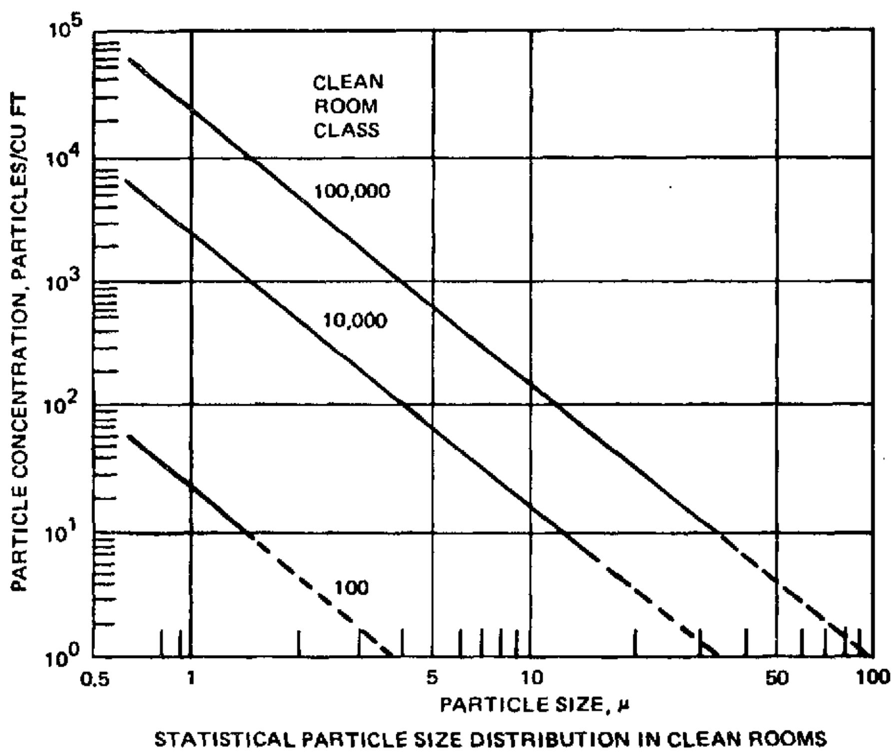

FIGURE C-1 CLEAN-ROOM STANDARDS

2.1 SHUTTLE INDUCED CONTAMINATION

The EOS will be mated to the Orbiter in the Orbiter Processing Facility (OPF). Little data is available on the characteristics of this facility or others used for payload inspection, checkout, propellant loading, pyro installation, and other operations prior to Orbiter mating. It is expected, however, that the OPF will be supplied with air at a Class 100,000 cleanliness level after the Orbiter is placed in the building. A hanging shroud will be placed over the open cargo bay and Class 5,000 air will be provided independently to this area.

Before loading the EOS/FSS, the cargo bay will be cleaned "to a visible clean level, as defined in JSC Spec. SN-C-0005". The cargo bay shroud area will continue to be purged until closeout with Class 5,000 air which contains less than 15 ppm hydrocarbons. The provision of air at this cleanliness level might be interpreted as suggesting the continued maintenance of a Class 5,000 environment within the shroud and bay. However, this is not specifically stated in JSC 07700, Volume XIV, and other sections tend to lead away from this interpretation. For example, paragraph 4.3.4.3 of that document (Preparation for Closeup of Payload Bay) states that prior to final closure, inspection and cleaning will be conducted as required to verify that all surfaces meet the visibly clean level criterion. The implication is that deposition of particulates can occur on surfaces within the Orbiter cargo bay, despite the continuous flow of very clean air.

After payload bay closure, the conditioned air purge is maintained up to 30 minutes prior to external tank propellant loading; at this time, the purge is continued with GN2 until lift-off. If the requirement exists to open the cargo bay doors at the launch pad, the operation will be conducted within an environmentally controlled payload changeout room. The potential incompatibility between the GN2-purged cargo bay and the air-purged changeout room has not been settled yet.

After launch, the Orbiter payload bay is vented and remains unpressurized until the reentry phase. As a design goal, overboard dumping of gases and liquids will be controlled to avoid contamination of the payload and payload bay. In addition, orbiter RCS thruster firings will be planned to avoid contamination when the payload bay doors are open. This is meant to include deployed or released payloads as well as the payload bay.

Table C-1 and C-2 summarize the sources and nature of the Orbiter originated contaminants expected during the various mission phases.

2.2 CONTAMINATION EFFECTS ON EOS

Many of the sensors to be flown in the EOS program will have 10,000 class cleanliness requirements. This environment will not be guaranteed by the Shuttle system; thus, we must consider the possible effects. Two effects are of major concern, the condensation of contaminants on optical surfaces and on radiative cooler surfaces. Condensation of contaminants on optical surfaces may be caused by direct (line-of-sight) impingement or indirect (reflected) impingement, either while in the Shuttle bay or during the process of deployment and checkout with the Shuttle Orbiter in the vicinity of the spacecraft. The condensible contributions of various effects are shown in Figure C-2 as a function of exposure duration.

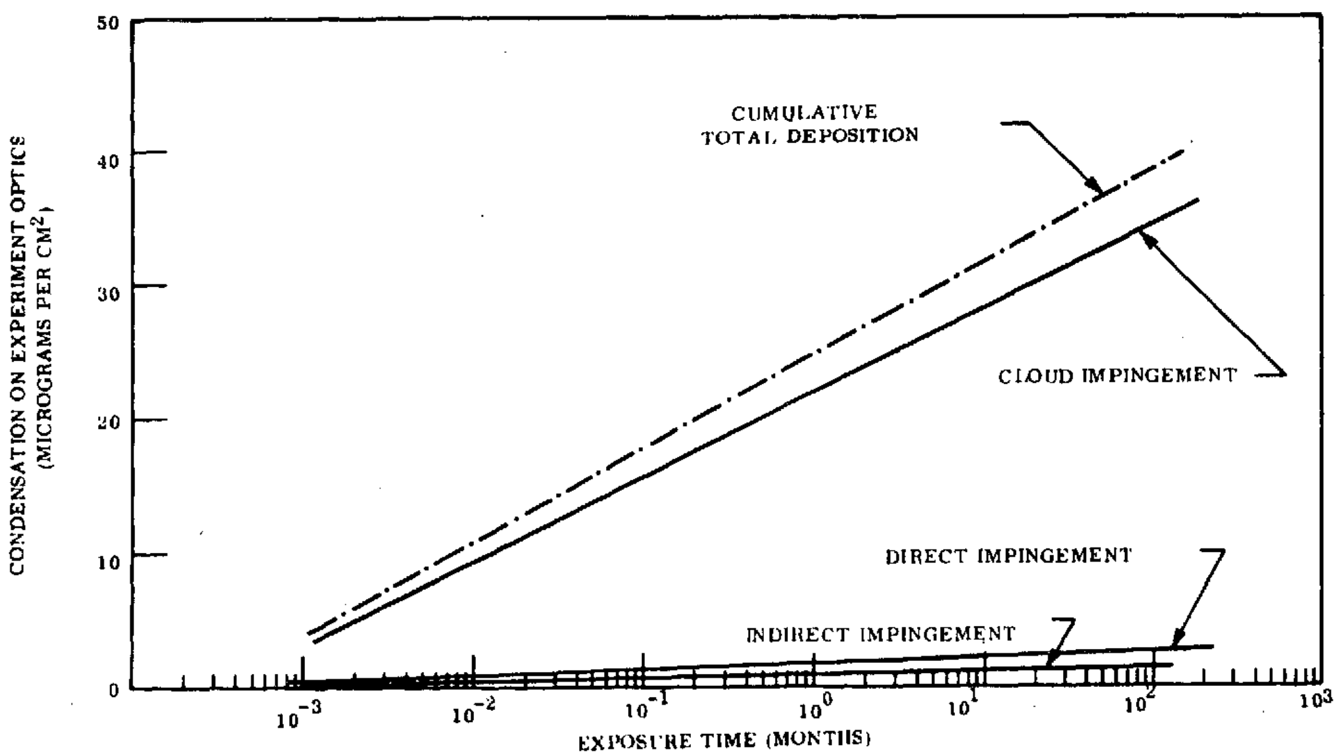

FIGURE C-2 CONDENSIBLE DEPOSITION VERSUS TIME

TABLE C-1 POTENTIAL CONTAMINATION SOURCES

MISSION PHASE	PAYLOAD EFFLUENT	STAGE EFFLUENT	SHUTTLE EFFLUENTS					
			RCS PLUME	OMS PLUME	LEAKAGE	EC/LS EFFLUENT	FUEL CELL PURGE	ABLATOR OUTGASSING
PRELAUNCH	LEAKAGE	POSSIBLE VENT LEAKAGE	N/A	N/A	N/A	LEAKAGE	N/A	N/A
LAUNCH PAD	LEAKAGE	VENT LEAKAGE	N/A	N/A - EVEN IF OMS USED P/L PROTECTED	POSSIBLE OMS OR ABES CONTAMINANTS	LEAKAGE	N/A	OUTGASSING - WILL NOT AFFECT P/L
ON ORBIT - BAY OPEN	LEAKAGE/ OUTGASSING	VENT LEAKAGE	N/A - AS LONG AS P/L IS NOT ERECTED	SAME AS RCS	SEE ABOVE	LEAKAGE/WASTE NO PROBLEM UNLESS HOLD TANKS FULL	SEE TABLE C-2	OUTGASSING EXPECTED TO LAST ONLY 24 HRS AT SIGNIFICANT LEVEL
P/L DEPLOYMENT	LEAKAGE/ OUTGASSING	VENT LEAKAGE	N/A - INHIBITED DURING DEPLOYMENT	N/A - NOT USED AT THIS TIME	SEE TABLE C-2 AND ABOVE	SEE TABLE C-2	SEE TABLE C-2	SEE ABOVE
SEPARATION	LEAKAGE/ OUTGASSING	VENT LEAKAGE	N_2, H_2, NH_3	N/A - SEE ABOVE	SEE ABOVE	SEE ABOVE	SEE ABOVE	SEE ABOVE
LOITER	LEAKAGE/ OUTGASSING	VENT LEAKAGE	N/A - DISTANCE	N/A - SEE ABOVE	N/A - DISTANCE	N/A - DISTANCE	N/A - DISTANCE	N/A - DISTANCE
SERVICE	LEAKAGE	VENT LEAKAGE	N_2, H_2, NH_3	N/A	SEE TABLE C-2 & OMS/ABES CONTAMINANTS	SEE TABLE C-2	SEE TABLE C-2	UNKNOWN

TABLE C-2 ORBITER EFFUENT DISCHARGE

SOURCE	EFFLUENT	RATE	OCCURRENCE
LEAKAGE			
HATCH	O_2, N_2	~0 TO 3.5 LB/DAY	CONTINUOUS
AVIONICS BAY	O_2, N_2 AND POTENTIAL EQUIPMENT OUTGASSINGS	~0 TO 400 STD. CC/HR	CONTINUOUS
EC/LS EFFLUENT			
WASTE (FECAL)	H_2O	0.02 LB/MAN-DAY	VENTED CONTINUOUSLY (LESS SHORT USE PERIODS)
MANAGEMENT ULLAGE	N_2, O_2	0.01 LB/CYCLE	2 CYCLES/MAN-DAY
	ATMOSPHERE AND METHANE		PURIFIED AND RECYCLED
FUEL CELL PURGE			
O_2 PURGE	O_2 A N_2 H_2O CO_2, HYDROCARBONS	0.03 - 0.07 LB 0.013 - 0.016 LB 0.01 LB 0.002 - 0.0075 LB TRACE	ONCE PER HOUR
ABLATOR	LARGE CARBON-SILICA MOLECULES AND HYDRO-CARBON GASES	APPROXIMATELY 0.5 TO 2 LB AFTER IN-SERTION	EACH MISSION; EXPONENTIALLY DECAYS; APPROX. 90% COM-PLETE 24 HRS. AFTER INSERTION

These depositions will affect the quality of sensor performance in two ways. First, by light absorption and second, by a loss of resolution due to light scattering by the deposited contaminants. At a maximum, the potential degradation could reach the levels shown in Table C-3.

TABLE C-3

ESTIMATED SENSOR PERFORMANCE DEGRADATION VERSUS SPECTRAL REGION

SPECTRAL REGION	WAVELENGTH (Å)	ABSORPTION DUE TO DEPOSITION (%)	LOSS OF RESOLUTION DUE TO DEPOSITION SCATTERING (%)
Far UV	800 - 1600	3	15
UV	1600 - 4000	3	30
Visible	4000 - 7000	2	30
Near IR	7000 - 15000	5	20
Far IR	$1.5 - 30 \times 10^3$	5	15

The effect of condensibles on instrument radiative coolers is to significantly reduce their efficiency, thus increasing the operating temperature of the detectors and degrading their performance. The likelihood of this occurring is relatively low since instrument radiators are typically covered or shielded, and are not at a significantly lower temperature than their surround until their exposure to cold space.

2.3 CONTAMINATION CONTROL/AVOIDANCE

If contamination of radiative coolers by condensibles cannot be avoided, the contaminants can be removed by periodic cleaning through evaporation. Heaters for this purpose may be incorporated into the cooler design and operated on ground command. This approach has been used effectively on the Surface Composition Mapping Radiometer (SCMR) on Nimbus E.

Contamination control/avoidance on sensor optics may be achieved by a number of design and operational countermeasures. The utilization of an Orbiter work shroud at the OPF during all open cargo bay operations and the maintenance of rigid clean room standards will help greatly to reduce the major source of contamination. Materials selection will obviously be controlled by the Shuttle Project Office, and this will also help.

For flight operations, there are a number of approaches which the Orbiter can use to limit contaminants. The most obvious is the control of Orbiter RCS/OMS thrusters, especially during periods of EOS deployment, separation, and retrieval. Without a high duty cycle use of the RCS, tight attitude control cannot be maintained by the Orbiter and an attached checkout of EOS mission sensors is limited; however, this is not a serious limitation.

Another critical operational method to avoid contamination is the avoidance and/or control of venting. Since it would be impractical to prohibit venting, appropriate control measures are required. Quantities can be made low and infrequent by proper design of vent ports, configuration, and duty cycle. All vent ports should be located away from critical areas and designed to provide high velocity, short-duration, directional flow. Designing tankage to provide minimum duty cycle is also desirable.

With optimum design and operational practices in the Shuttle program, contamination impact on EOS instruments can be minimized. If needed, additional countermeasures can be introduced. Perhaps the simplest of these would involve a simple purge system. Another approach is the design of a protective shroud for the spacecraft. One concept recently developed by GE and MDAC as part of the Payload Utilization of Tug (PUT) study is shown in Figure C-3. This design concept is adaptable to the EOS program, but would require a significant change in the FSS. Obviously the shroud should be retained in the Orbiter cargo bay after deployment, and not be permanently attached to the spacecraft.

In addition to the design countermeasures proposed, the avoidance of contamination in flight may be aided by careful interleaving of EOS checkout, deployment, and separation operations with those of the Orbiter. The coordination of RCS thruster firings and planned liquid and gaseous venting with the elevation and release of the EOS should significantly reduce the chance of serious contamination of sensor equipment.

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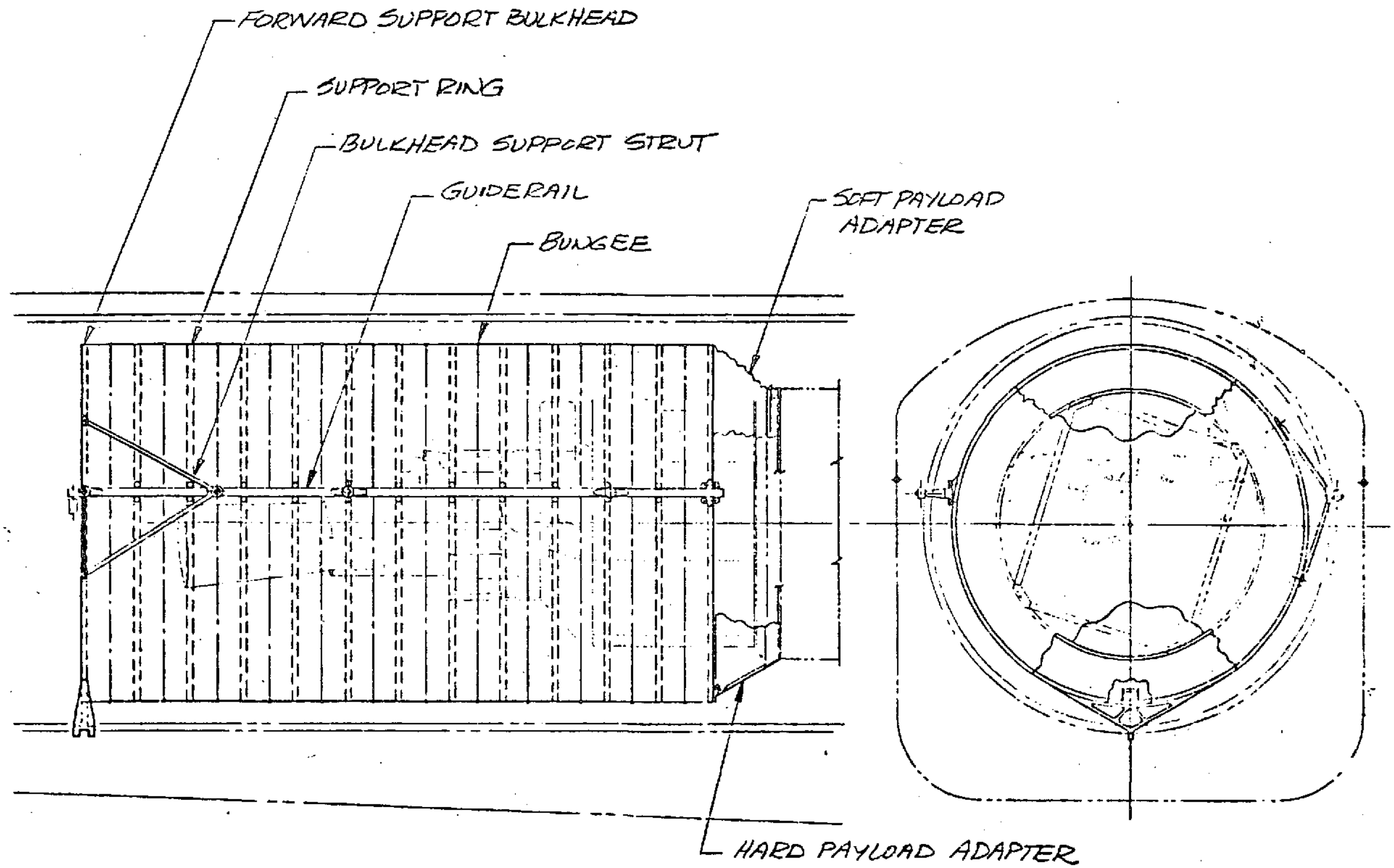

FIGURE C-3
"ACCORDIAN" PAYLOAD SHROUD CONCEPT (PUT STUDY)

3.0 THERMAL CONTROL

3.1 SHUTTLE THERMAL ENVIRONMENT

The Orbiter bay thermal environments are defined in the Space Shuttle System Payload Accommodations Report, JSC 07700 (Volume XIV, Revision C) and shown in the following tables and figures. Table C-4 presents the maximum range payload bay wall thermal environments while Figures C-4 through C-7 show payload bay liner temperature as a function of:

- o Location on payload bay liner
- o time
- o payload heat sink.

A curve of estimated EOS heat sink has been superimposed on each of these figures. The figures provide more realistic estimates of the actual thermal environments that will be experienced by EOS in the Shuttle bay. Note that the temperatures presented are only for use "as a guide" to initiate the thermal design and integration and should be followed by detailed integrated analysis between Shuttle and the payload as required.

TABLE C-4 PAYLOAD BAY WALL THERMAL ENVIRONMENT

CONDITION	DESIGN MINIMUM	DESIGN MAXIMUM
Prelaunch	+40°F (4.5°C)	+120°F (49°C)
Launch	+40°F (4.5°C)	+150°F (65.5°C)
On-Orbit (doors closed)	See C&D	See A&E
Entry and postlanding	-100°F (-73°C)	+200°F (93.5°C)

Heat leak criteria into or out of a 100°F (37.5°C) constant payload are as follows:

A. Total bay heat gain, average	$\leq$ 0 Btu/Ft^2-hr	(0 Watt/Meter^2)
B. Heat gain, local area	$\leq$ 3 Btu/Ft^2-hr	(9.5 Watt/Meter^2)
C. Total bay heat loss, average	$\leq$ 3 Btu/Ft^2-hr	(9.5 Watt/Meter^2)
D. Heat loss, local area	$\leq$ 4 Btu/Ft^2-hr	(12.8 Watt/Meter^2)

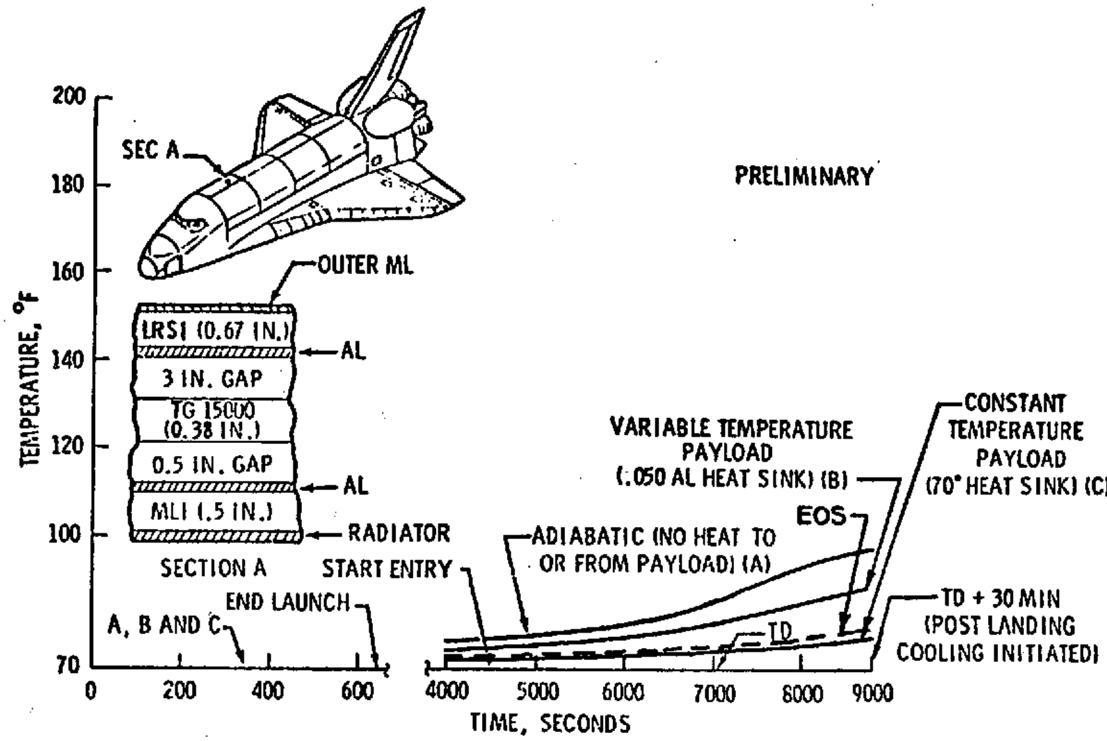

FIGURE C-4
PAYLOAD BAY LINER TEMPERATURE (TOP C_L X/L = .3)

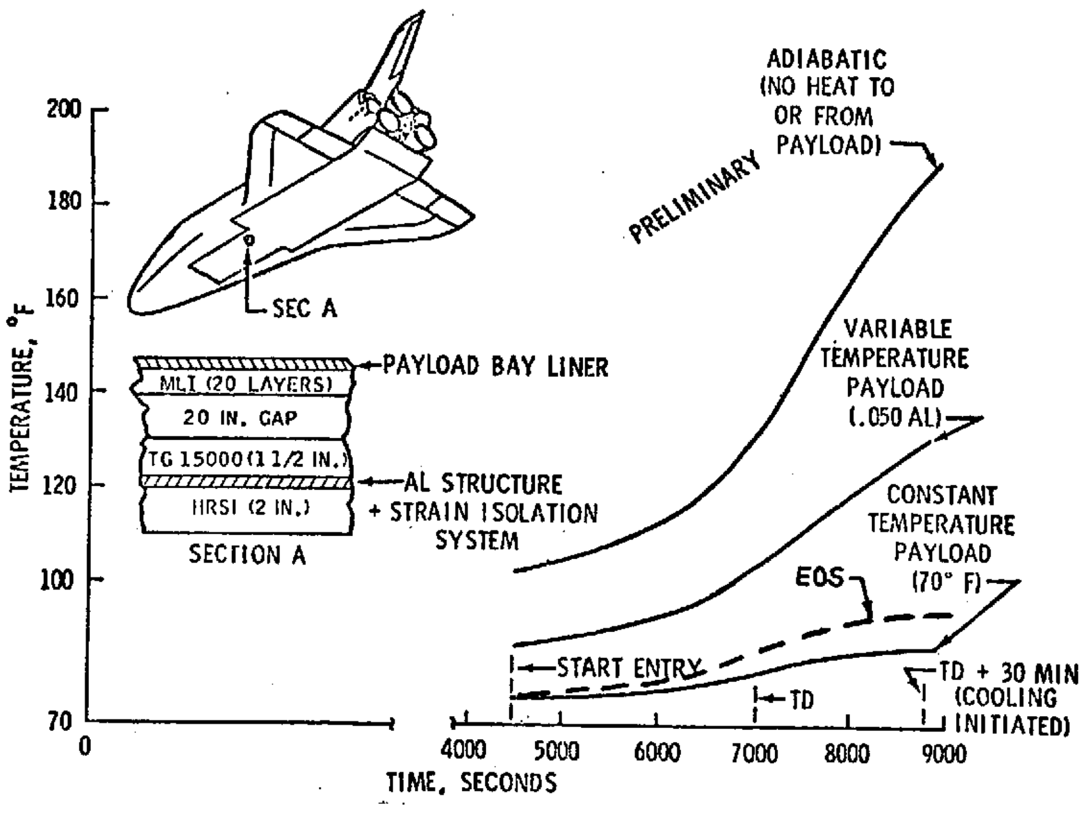

FIGURE C-5
PAYLOAD BAY LINER TEMPERATURE (BOTTOM, FWD EQUIP BAY)

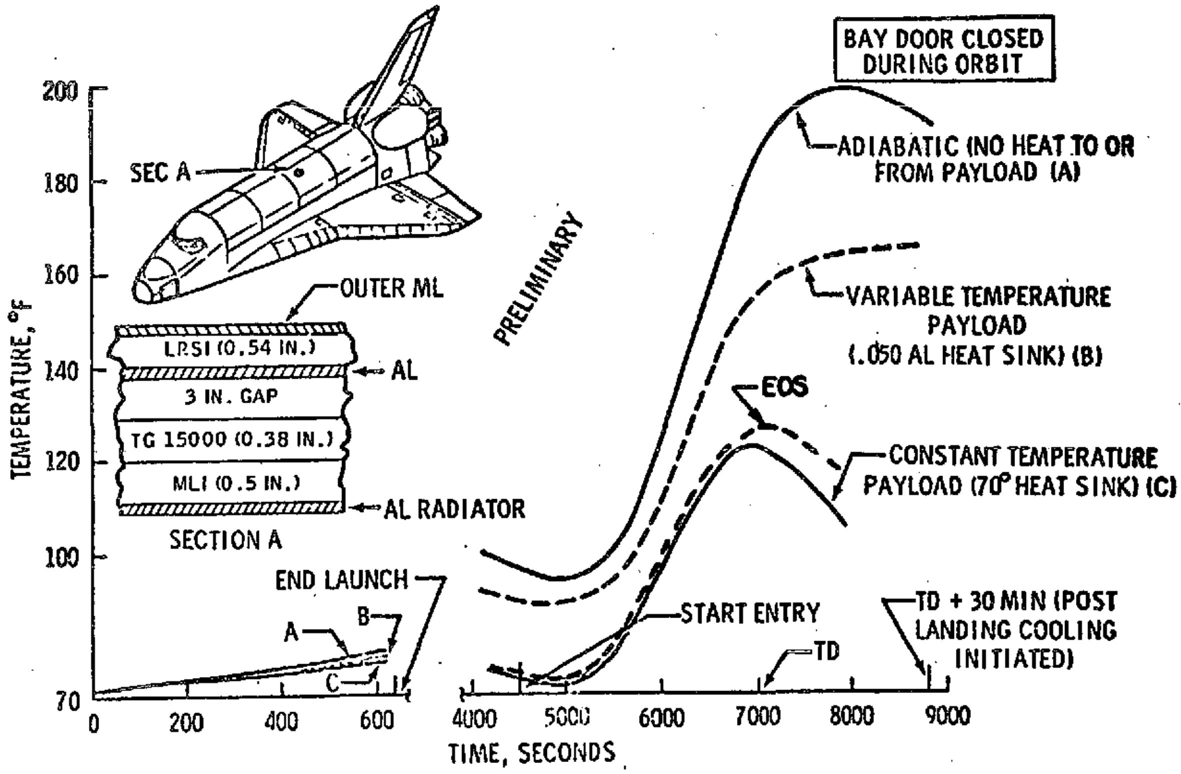

FIGURE C-6

PAYLOAD BAY LINER TEMPERATURE (TOP C_L X/L = .6)

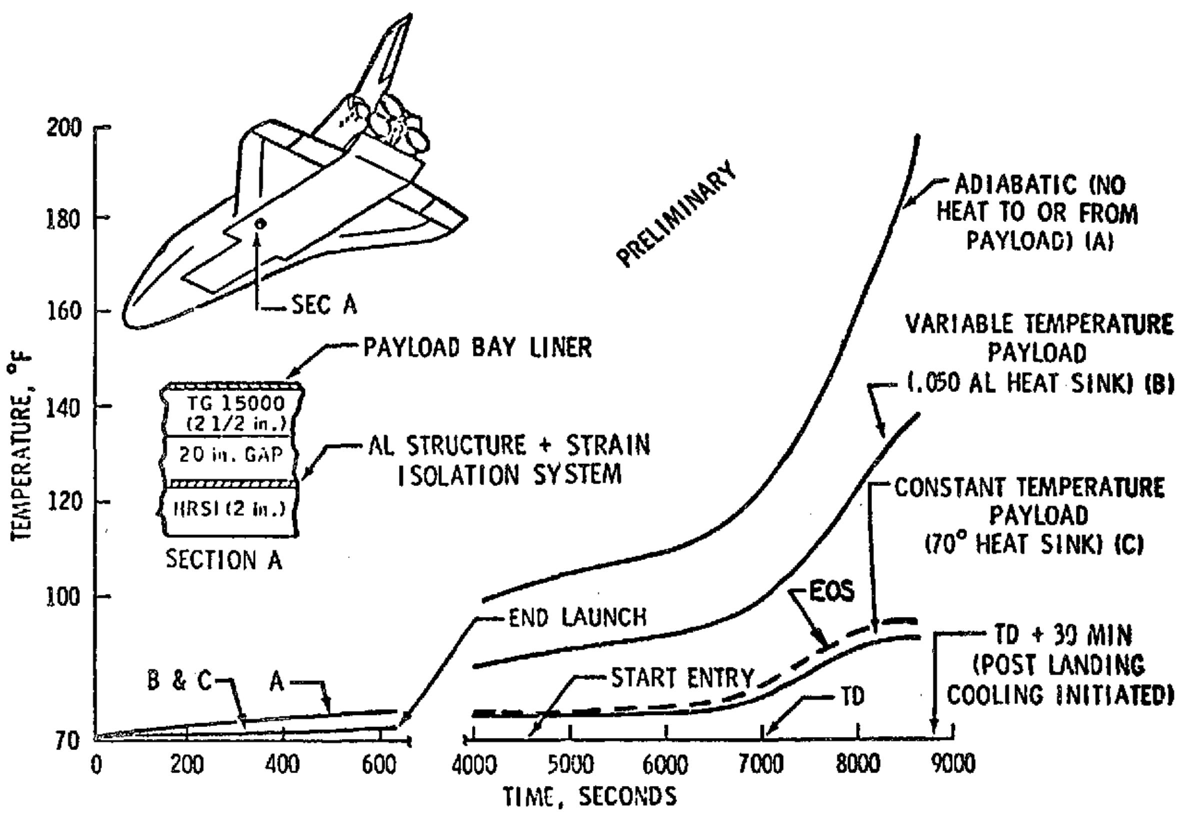

FIGURE C-7

PAYLOAD BAY LINER TEMPERATURE (BOTTOM, AFT OF FWD EQUIP BAY)

3.2 PRELAUNCH THERMAL CONDITIONS

During prelaunch the maximum air flow temperature (potential range 45 to 120°F) should be restricted to below 86°F (30°C), the maximum allowable non-operating temperature limit for the batteries. EOS spacecraft temperature control during launch can also be simplified by maintaining a relatively cool prelaunch environment in the Shuttle bay. The establishment of an initial cool environment at lift off, and the inherent thermal capacitance of the batteries allows their temperature to be maintained below 95°F/35°C (the maximum allowable battery transient temperature) during launch without impacting the passive EOS spacecraft thermal design. Thus it can be seen that there are significant benefits to limiting the prelaunch air flow temperature below 86°F (30°C).

3.3 ON-ORBIT THERMAL CONDITIONS

The basic spacecraft passive thermal design (established for on-orbit performance) is capable of maintaining temperatures during on-orbit (door closed) conditions with the average and local bay heat gains defined in Table C-4. Shuttle power available through the spacecraft umbilical will be used to maintain module minimum temperatures for the bay heat losses defined in Table C-4. Total Shuttle power of up to 50.5 watts will be required for the minimum conditions. The distribution of this power as a function of spacecraft module or component is summarized in Table C-5.

TABLE C-5
ON-ORBIT EOS POWER REQUIREMENTS (DOOR CLOSED, MINIMUM ENVIRONMENT)

MODULE OR COMPONENT	SHUTTLE POWER REQUIRED - WATTS
ACS MODULE	3.6
G& DH MODULE	8.6
POWER MODULE	4.6
RCS ENGINES & VALVES	4.0
O.A./O.T. ENGINES & VALVES	7.8
TANK/LATCHING VALVE/LINES	5.4
SOLAR ARRAY DRIVE	1.5
INSTRUMENTS	15.0
TOTALS	50.5

During periods with the door open the maximum heater power required will depend on the minimum values of external heat flux (presently undefined in Shuttle documentation). At this point, it is assumed that the total heater power required will not exceed the 50.5 watts previously defined.

3.4 ENTRY AND POST-LANDING

During entry and post-landing (+30 minutes), the minimum average design temperature of -100°F would cause an excessive heater power penalty. However, the thermal time constant of the EOS spacecraft modules is about 2.3 hours, indicating that the vehicle could survive the anticipated environment using only its time constant and local heater power of about 18.7 watts total. The local heater power supplied by the Orbiter would be used for components such as rocket engine valves and lines and the solar array drive which have smaller thermal time constants. The maximum environment of 200°F shown in Table C-4 is shown to be very pessimistic when presented as shown in Figures C-4 through C-7, as a function of time, location on the payload bay liner and payload heat sink. Utilizing the EOS spacecraft time constant (see EOS curves superimposed on Figures C-4 through C-7) the maximum temperature is shown to be well below its 95°F non-operating transient limits in three of the four Orbiter bay liner locations defined. The one area where temperatures exceed the limit of 95°F is toward the aft end of the Orbiter bay along the top centerline. Present layouts of the EOS spacecraft mounted in the bay (shown in Section 2 of this report) indicate that the critical components (batteries) can be located away from this local hot spot. It should be noted that the average wall temperature will be well below its 95°F requirement, and the actual EOS temperatures will lag those shown for the bay wall.

Specific interface requirements with the Shuttle will be further refined as both the Shuttle and EOS designs mature. However, with the preliminary data available on Shuttle, the EOS design appears compatible.